ORBITAL TRANSFER VEHICLE

CONCEPT DEFINITION AND SYSTEMS ANALYSIS STUDY

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Volume II, Book 2

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FOREWORD

This final report of the Orbital Transfer Vehicle (OTV) Concept Definition and System Analysis Study was prepared by Boeing Aerospace Company for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center in accordance with Contract NAS8-36107. The study was conducted under the direction of the NASA OTV Study Manager, Mr. Donald Saxton and during the period from 1984 to September 1986.

This final report is organized into the following nine documents:

VOL. I	Executive Summary	
VOL. II	OTV Concept Definition & Evaluation	
	Book 1 -	Mission Analysis & System Requirements
	Book 2 -	Selected OTV Concept Definition - Phase I
	Book 3 -	Configuration and Subsystem Trade Studies
	Book 4 -	Operations and Propellant Logistics
VOL. III	System & Program Trades	
VOL. IV	Space Station Accommodations	
VOL. V	WBS & Dictionary	
VOL. VI	Cost Estimates	
VOL. VII	Integrated Technology Development Plan	
VOL. VIII	Environmental Analysis	
VOL. IX	Implications of Alternate Mission Models and Launch Vehicles	

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ABBREVIATIONS AND ACRONYMS

ACC	Aft Cargo Carrier
AFE	Aeroassist Flight Experiment
AGE	Aerospace Ground Equipment
AL	Aluminum
ASE	Airborne Support Equipment
A/T	Acceptance Test, Auxiliary Tank
AUX	Auxiliary
AVG	Average
B/B	Ballute Brake
B/W	Backwall
CDR	Critical Design Review
CPU	Central Processing Unit
CUM	Cumulative
DAK	Double Aluminized Kapton
DDT&E	Design, Development, Test & Evaluation
DELIV	Delivery
DMU	Data Management Unit
DoD	Department of Defense
EPS	Electrical Power System
FACIL	Facility
FFC	First Flight Certification
FLTS	Flights
FOSR	Flexible Optical Surface Reflector
FRCI	Fiber Refractory Composite Insulation
F.S.	Fail Safe
FSI	Flexible Surface Insulation
FTA	Facilities Test Article
GB	Ground Based
GEO	Geostationary Earth Orbit
GPS	Global Positioning System
GRD	Ground
IOC	Initial Operational Capability
IRU	Inertial Reference Unit
IUS	Inertial Upper Stage

JSC	Johnson Space Center
L/B	Lifting Brake
LCC	Life Cycle Cost
L/D	Lift to Drag
MGSS	Mobile GEO Service Station
MLI	Multilayer Insulation
MPS	Main Propulsion System
MPTA	Main Propulsion Test Article
MSFC	Marshall Space Flight Center
OMV	Orbital Maneuvering Vehicle
OPS	Operations
OTV	Orbital Transfer Vehicle
PAM	Payload Assist Module, Propulsion Avionics Module
PDR	Preliminary Design Review
PFC	Preliminary Flight Certification
P/L	Payload
PROD	Production
PROP	Propellant
RCS	Reaction Control System
REF	Reference
RGB	Reusable Ground Based
R&R	Remove & Replace
RSB	Reusable Space Based
RSI	Reusable Surface Insulation
SB	Space Based
S/C	Spacecraft
SCB	Shuttle Cargo Bay
SIL	Systems Integration Laboratory
STA	Structural Test Article
STG	Stage
STS	Space Transportation System
T/D	Turndown
TDRS	Tracking Data Relay Satellite
TPS	Thermal Protection System
TT&C	Telemetry, Tracking and Control
WBS	Work Breakdown Structure

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1.0 INTRODUCTION

This section provides an overview of the entire study effort in terms of background, objectives and issues, study/report organization, and content of this specific volume.

Use of trade names, names of manufacturers, or recommendations in this report does not constitute an official endorsement, either expressed or implied, by the National Aeronautics and Space Administration.

And finally, it should be recognized that this study was conducted prior to the STS safety review that resulted in an STS position of "no Centaur in Shuttle" and subsequently an indication of no plans to accommodate a cryo OTV or OTV propellant dump/vent. The implications of this decision are briefly addressed in section 2.2 of the Volume I and also in Volume IX reporting the Phase II effort which had the OTV launched by an unmanned cargo launch vehicle. A full assessment of a safety compatible cryo OTV launched by the Shuttle will require analysis in a future study.

1.1 BACKGROUND

Access to GEO and earth escape capability is currently achieved through the use of partially reusable and expendable launch systems and expendable upper stages. Projected mission requirements beyond the mid-1990's indicate durations and payload characteristics in terms of mass and nature (manned missions) that will exceed the capabilities of the existing upper stage fleet. Equally important as the physical shortfalls is the relatively high cost to the payload. Based on STS launch and expendable upper stages, the cost of delivering payloads to GEO range from $12,000 to $24,000 per pound.

A significant step in overcoming the above factors would be the development of a new highly efficient reusable upper stage. Numerous studies (ref. 1, 2, 3, 4) have been conducted during the past decade concerning the definition of such a stage and its program. The scope of these investigations have included a wide variety of system-level issues dealing with the type of propulsion to be used, benefits of aeroassist, ground- and space-basing, and impact of the launch system.

1.2 OBJECTIVES AND ISSUES

The overall objective of this study was to re-examine many of these same issues but within the framework of the most recent projections in technology readiness, realization that a space station is a firm national commitment, and a refinement in mission projections out to 2010.

The output of this effort was twofold: (1) the definition of a preferred OTV concept(s) and its programmatics, and (2) definition of the key interfaces that would occur between the OTV and space station.

During the nineteen-month technical effort the specific issues addressed were:

a. What are the driving missions?
b. What are the preferred space-based OTV characteristics in terms of propulsion, aeroassist, staging, and operability features?
c. What are the preferred ground-based OTV characteristics in terms of delivery mode, aeroassist, and ability to satisfy the most demanding missions?
d. How extensive are the orbital support systems in terms of propellant logistics and space station accommodations?
e. Where should the OTV be based?
f. How cost effective is a reusable OTV program?
g. What are the implications of using advanced launch vehicles?

1.3 STUDY/REPORT ORGANIZATION

Accomplishment of the objectives and investigation of the issues was done considering two basic combinations of mission models and launch systems. Phase I concerned itself with a mission model having 145 OTV flights during the 1995-2010 timeframe (Revision 8 OTV mission model) and relied solely on the Space Shuttle for launching. Phase 2 considered a more ambitious model (Rev. 9) having 442 flights during the same time frame as well as use of a large unmanned cargo launch vehicle and an advanced Space Shuttle (STS II).

The study is reported in nine separate volumes. Volume I presents an overview of the results and findings for the entire study. Volume II through VIII contains material associated only with the Phase I activity. Volume IX presents material unique to the Phase II activity. Phase I involved five quarters of the technical effort and one quarter was associated with the Phase II analyses.

1.4 DOCUMENT CONTENT

This specific document reports the work associated with the recommended OTV concept resulting from the Phase I effort. The key program groundrules influencing this selected concept was the Revision 8 mission model and use of an STS with 72K-lbm lift capability. System level trade studies contributing to selection of a reusable ground-based concept are presented in Volume III and vehicle level analysis is discussed in Volume II, Book 3. The remainder of this document will discuss overall system design and operational characteristics, subsystem features, performance, and programmatics associated with the selected concept. Unless otherwise stated, the units used in this document are English.

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2.0 SYSTEM DESCRIPTION

This section presents an overall description of the selected ground-based, ballute-braked OTV concept and includes a presentation of the overall system requirements, and a top-level description of the OTV main stage and auxiliary tank systems, including performance, and launch and recovery operations.

2.1 SYSTEM LEVEL REQUIREMENTS

The key design requirements for the OTV system are given in table 2.1-1. The requirements shown are those which primarily affect the flight system.

2.2 SYSTEM DESIGN AND OPERATION

The selected Orbital Transfer Vehicle (OTV) is a ground-based cryogenic (LO_2/LH_2), ballute-braked stage carried to low earth orbit fully fueled within the payload bay of the Space Shuttle Orbiter. The selected system consists of the OTV main stage, with associated airborne support equipment, and a reusable auxiliary tank module with associated airborne support equipment.

For unmanned GEO delivery payloads of up to 12,000 lbs, the fully-fueled OTV main stage and payload are launched together in a single orbiter launch. For larger or more demanding payloads, the fully fueled OTV main stage and fueled auxiliary tank module with payload attached are launched in separate orbiter launches, and then mated at the Space Station. For delivery payloads of up to 12,000 lbs, a 33 ft diameter expendable ballute is carried on the OTV for use as an aerobrake. For other delivery missions, a 42 ft diameter ballute is needed because of the additional weight of the auxiliary tank module. For manned sortie missions, the OTV is configured for a manned flight by adding redundancy to key components, and 66 ft diameter ballute is needed to return the manned service cab, as well as the auxiliary tank set.

2.3 MAIN STAGE

The configuration of the selected ground-based OTV main stage is shown in figure 2.3-1. The main stage is made up of the following subsystems:

- o Structural - includes an external, load bearing body shell, an avionics/equipment support ring, a LH_2 tank, a LO_2 tank, and a ballute-type aeroassist device with support structure.
- o Propulsion systems - main propulsion consists of 2 advanced expander cycle engines with extendable nozzles, electromechanical actuator thrust vector control, and propellant delivery, pressurization, and vent systems; attitude

TABLE 2.1-1
OTV CONFIGURATION REQUIREMENTS

- General
  - Reusability—All vehicles to be designed to be retrieved and refurbished
    - Airframe - 40 mission service life
    - Tankage - 40 mission service life
    - Avionics - 40 mission service life
    - Aeroassist - 1 mission life for ballute

      5 mission life for lifting brake

      20 mission life for shaped brake
    - Main Engine (ASE) - 10 hours, 20 flights
    - On-Orbit
      Storage Tanks - 5 year service life
  - Satisfy Safety Requirements
    - Shuttle/Space Station
    - OTV Mission - No single credible failure shall preclude the safe return of the crew
  - Any hardware jettisoned during a mission shall be disposed of through controlled deorbit or other acceptable non-interference mode
  - OTV System shall be NASA STDN and TDRS compatible (communications and tracking)
  - The OTV design shall include the following flight performance reserves:
    - Main propulsion - 2% on each delta-V maneuver

TABLE 2.1-1 (continued)
OTV CONFIGURATION REQUIREMENTS

- o Reaction control system - 10% of mission nominal RCS propellant
- o Electrical power system - 20% of mission nominal reactants

o Mission Times - Use 12 hours at LEO for phasing

- o GEO delivery - 1 day at GEO
- o Manned GEO sortie - 18 days at GEO

o Pre-Launch

o Ground services (electrical, fluid, and gases) will be through orbiter service panels

o Launch

o The OTV and its payload will be launched to orbit by the STS, either in the Orbiter cargo bay or in the aft cargo carrier (ACC).

o The sum of the masses of the OTV and its consumables, the airborne support equipment and its consumables, orbiter-furnished airborne support equipment, and payload shall not exceed the weight determined by the following:

Launch wt = 87,960 - 114 (altitude, in nm)

o The OTV system shall provide for a structural adaptor and a deployment/release mechanism.

o Satisfy the static and dynamic loads, thermal, contamination, physical envelope, CG, and other requirements of payload accommodations handbook, Vol XIV of JSC document 07700.

o The OTV system shall provide for the dumping of propellants through the orbiter service panels in the event of an abort.

o Mission - Significant Payloads

o 20,000 lb delivery to GEO limited to 0.1 g max. acceleration

TABLE 2.1-1 (continued)
OTV CONFIGURATION REQUIREMENTS

- o 10,000 lb multiple-manifest payload to GEO
- o 7500 lb GEO manned sortie with 7500 lb return

o Recovery

- o (Space-Based) retrieved by OMV from parking orbit - OTV to remain passive during docking and reberthing at space station
- o (Ground Based) retrieved by shuttle RMS from parking orbit
  - o OTV to remain passive during docking and reberthing
  - o Reconnect umbilicals for purge and status monitoring prior to reentry

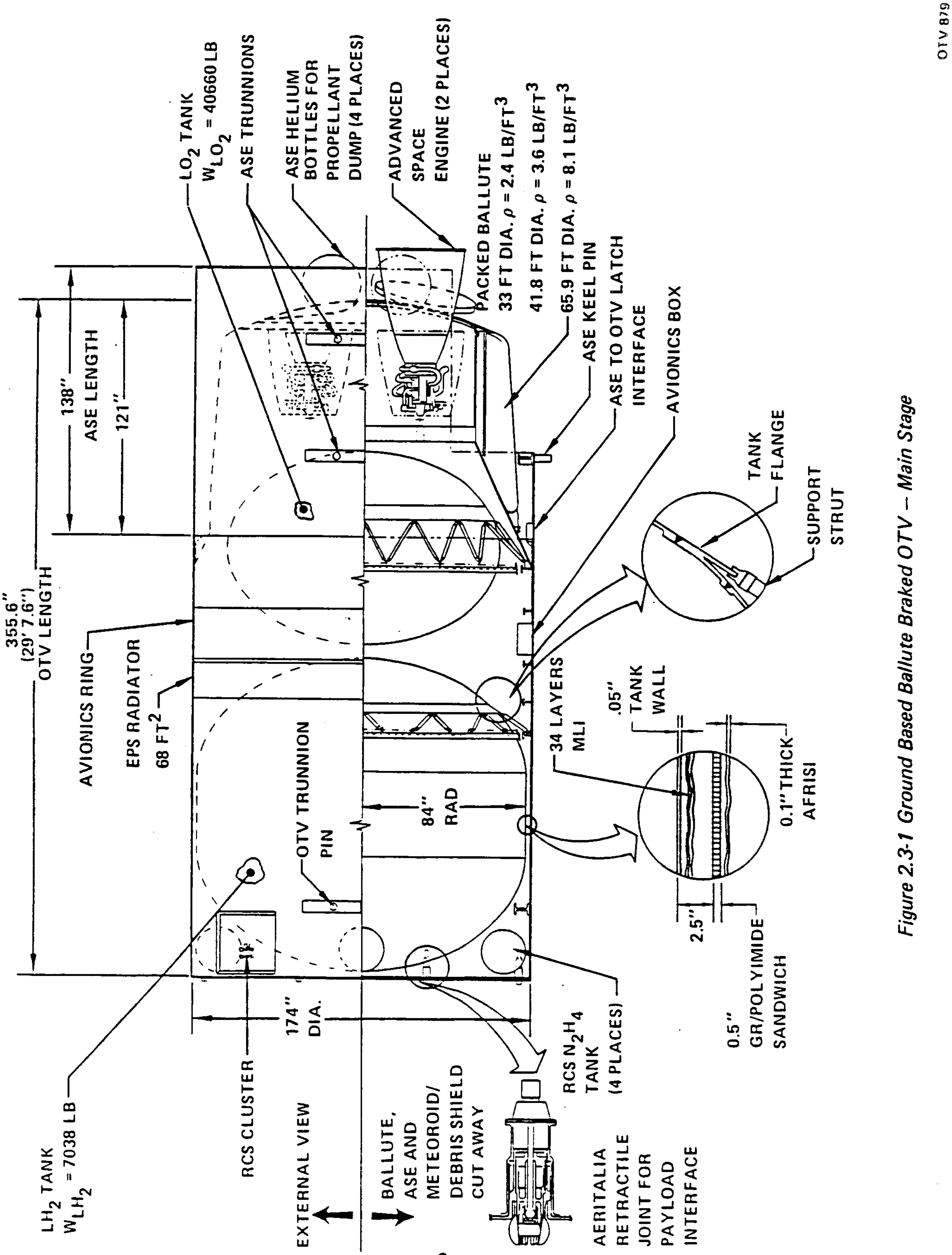

Figure 2.3-1 Ground Based Ballute Braked OTV – Main Stage

control consists of four hydrazine thruster modules and associated tankage, pressurization, and control.

- o Thermal control - both active and passive to regulate heating loads for overall mission and for aeromaneuver.
- o Guidance and Navigation - necessary redundancy for unmanned mission, with provisions (wiring, fittings) for upgrading to man-rated capability.
- o Communications and Data Handling - necessary redundancy for unmanned missions, with provisions (wiring, fittings, larger boxes) for upgrading to man-rated capability.
- o Electrical Power - features redundant O_2/H_2 fuel cells and distribution and control units, with provisions (wiring, fittings) for upgrading to man-rated capability.

The airborne support equipment (ASE) is that portion of the OTV system flight hardware which remains in the Shuttle Orbiter payload bay when the OTV is deployed. The ASE is a structural system that also provides for electrical, fluid, and avionics interfaces between the OTV and the Orbiter and provides tilt capability for OTV deployment. It also provides for abort dump pressurization.

A more detailed description of vehicle subsystems is given in the following paragraphs.

Structures. This group consists of the vehicle body structure, main propellent tankage, and aeroassist device structures.

Body Structure. The main propellent tanks are supported by struts within an external graphite/epoxy honeycomb sandwich shell. The shell is divided into 5 sections, one of which has aluminum mounting plates for installation of avionics and electrical power subsystems. Another section supports the thrust structure, upon which are mounted the two main engines. Major rings are located at each tank support location, as well as at the payload interface, ASE interface, and the ballute support structure/thrust structure interface. Secondary structures include payload support mechanisms, and orbiter handling and attachment fittings.

Avionics/Equipment Support Ring. The avionics/equipment support ring is a circular GR/EP structure with aluminum doors for mounting of avionics and electrical power components. The layout of the components is shown in figure 2.3-2. Equipment associated with particular subsystems have been located together. The first four bays

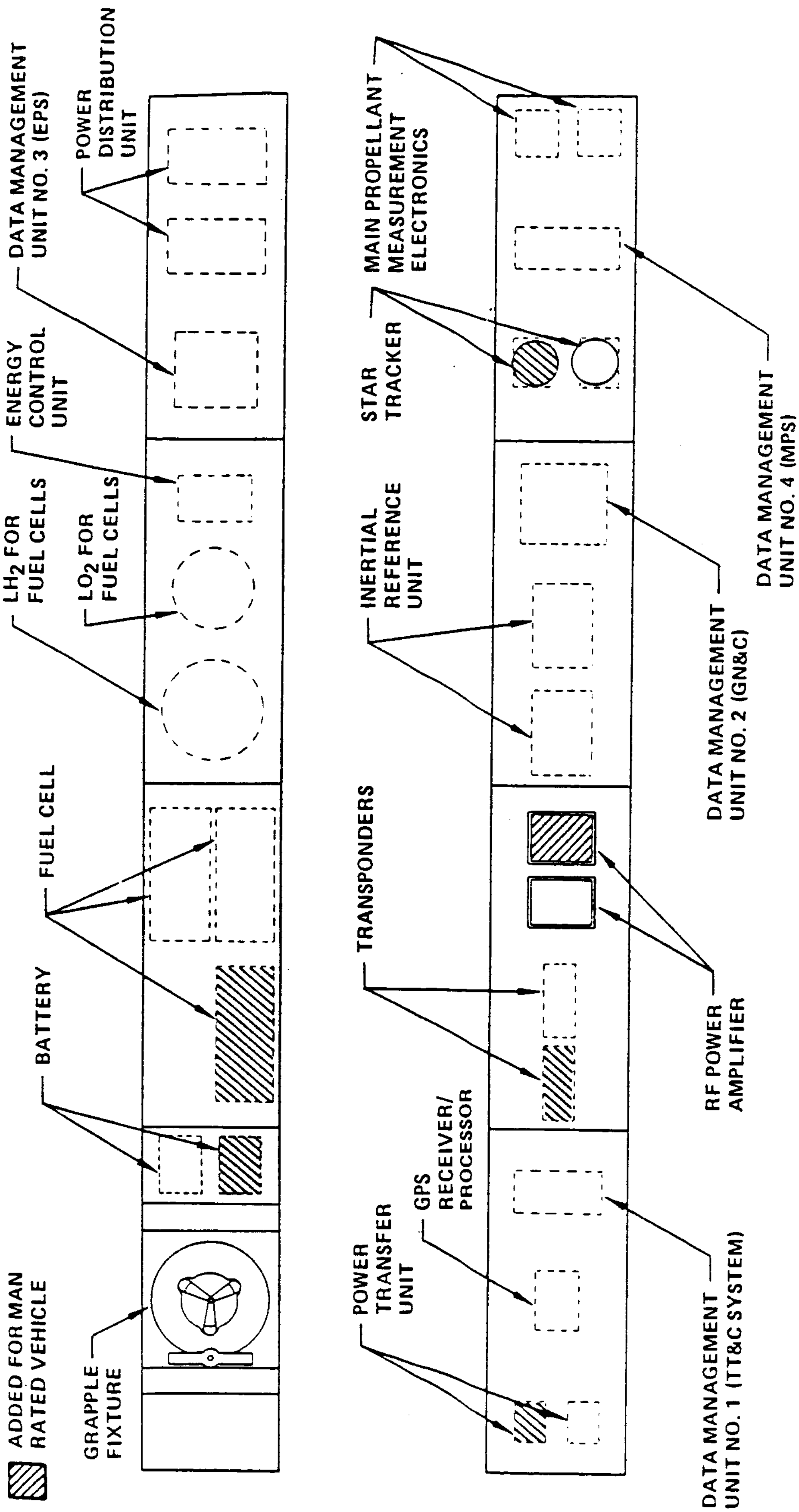

Figure 2.3-2 Avionics/Equipment Ring

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and part of the fifth are dedicated to the electrical power system. The fifth and sixth bays contain the TT&C subsystem. The seventh and part of the eighth contain the GN&C subsystem and the main propulsion subsystem occupies the remainder of the bay. Provisions needed to man-rate the vehicle are initially installed, but the necessary components themselves are only installed for the manned flights.

Main Propellant Tanks. The LH_2 and LO_2 tanks are all-welded 2219-T87 aluminum and are supported by struts within the external body shell. Both tanks have zero-g start baskets for propellant acquisition and are sized to a usable propellant capacity of 46,460 lbs at an oxidizer-to-fuel mixture ratio of 6:1. The propellant tank shells are designed to permit room temperature proof testing to ensure service life requirements. The hydrogen tank has 0.707 elliptical heads and a cylindrical barrel section. The average shell thickness in the heads is 0.030 in and the average barrel skin thickness is 0.05 in. The oxygen tank has 0.707 elliptical heads with an average skin thickness of 0.031 in.

Aeroassist Device. The aeroassist device consists of a jettisonable ballute and support structure, as well as reusable support structure and inflation provisions. The ballutes available for this system are 33 ft, 42 ft, and 66 ft diameters, as shown in figure 2.3-3. The ballutes are made of Nextel fabric gores sewn together and attached to the support structure at forward and aft attach points. Meridian straps, also of Nextel, carry tension loads to the attach points. Jettisonable support structure includes a ballute attachment ring and a graphite/epoxy ballute support cylinder with clamp and installation provisions. Ballute installation provisions and structures are shown in figure 2.3-4. The system is designed to ease installation of the ballute, keep the chance of damage to the heat shield during installation to a minimum, and assure jettisoning of the ballute after use. During installation, rollers on the ballute structure engage guide tracks on the heat shield support structure. After initial positioning, the ballute structure is winched onto the vehicle, then is seated and clamped. The forward attach point of the ballute is then threaded forward and seated into the forward attach clamp. The ballute attach points, as well as the ballute jettison sequence following the aeromaneuver, are shown in figure 2.3-5. These structures are detailed further in Vol. II, Book 3, Section 2.2.1 and 3.3.1. The reusable support structure includes a graphite/epoxy honeycomb sandwich support shell and a graphite/polyimide sandwich heat shield with structural doors for main engine thermal protection. Inflation provisions include KEVLAR-overwrapped GN_2 pressurant bottles, plumbing, and control hardware for ballute pressurant.

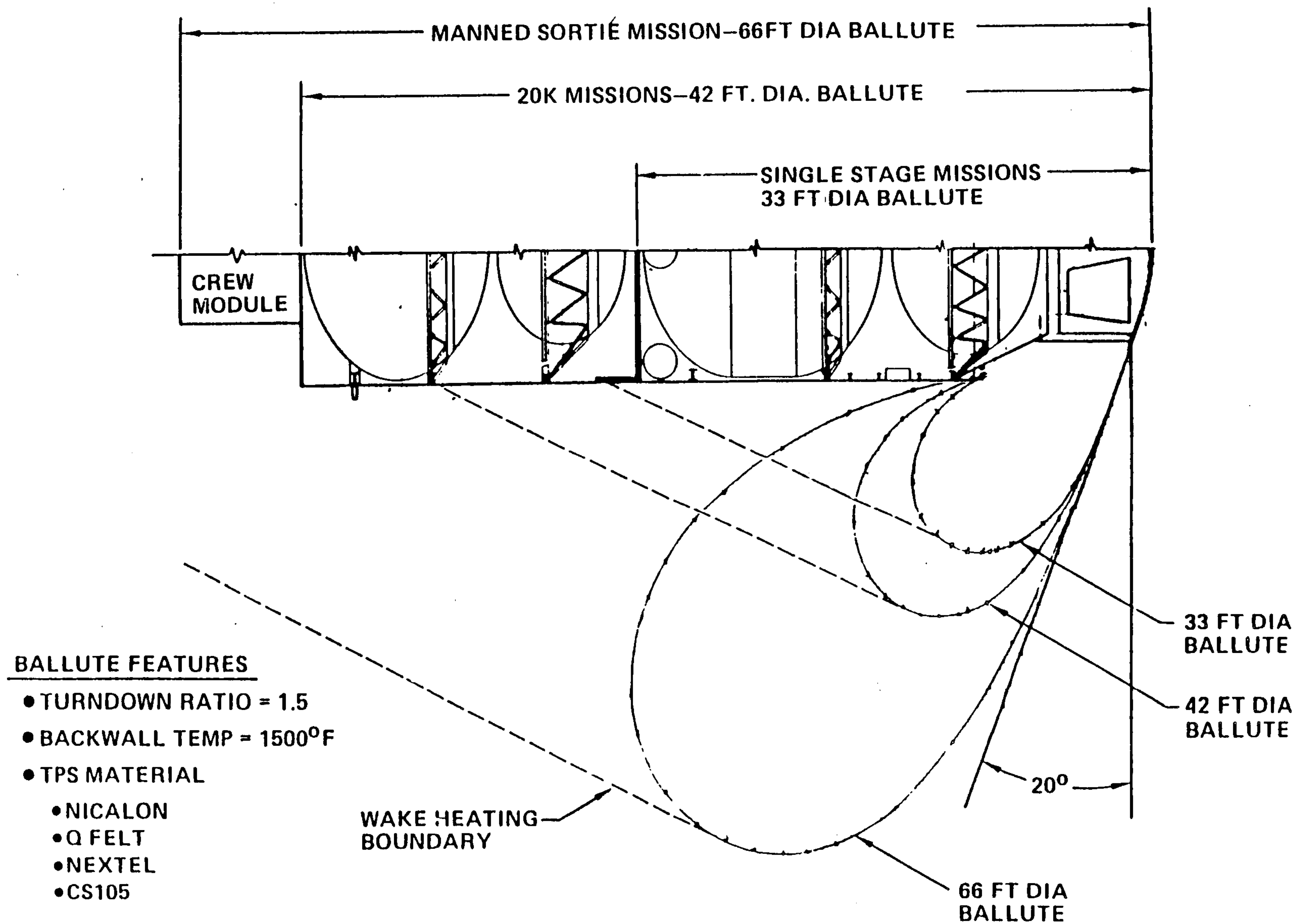

Figure 2.3-3 Ballute Aeroassist Provisions – Ground Based OTV

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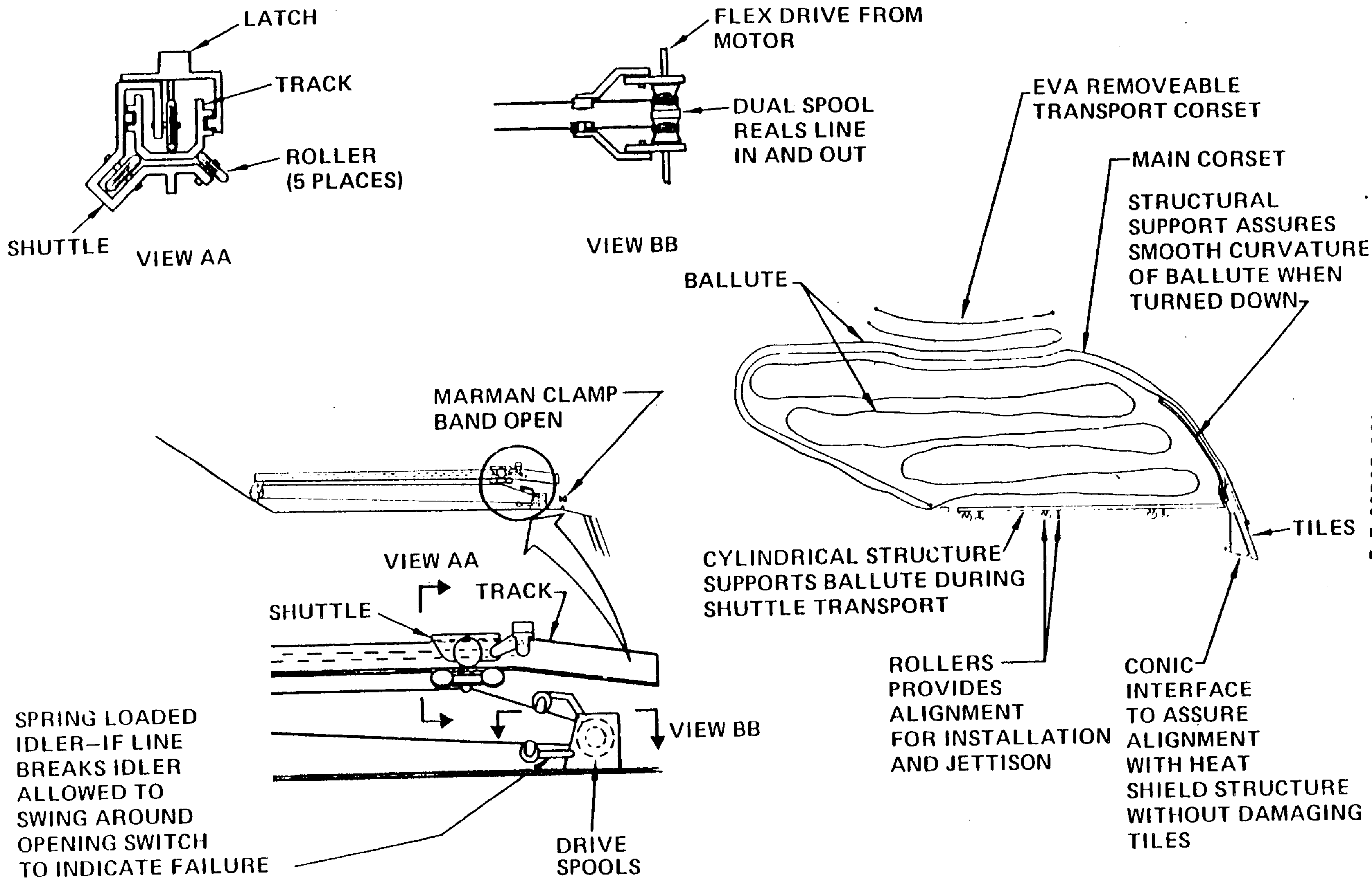

Figure 2.3-4 Ballute Installation/Jettison Provisions

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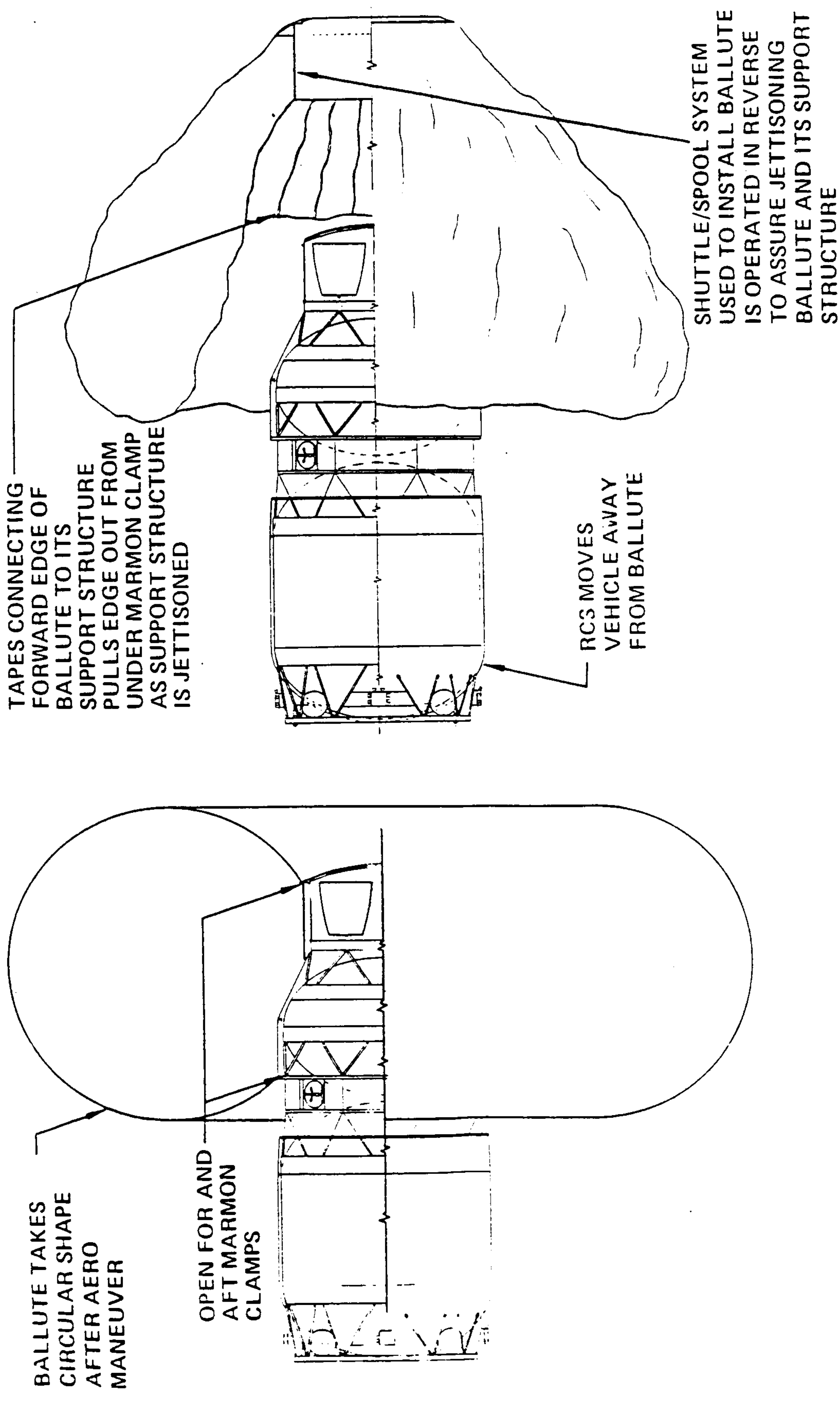

Figure 2.3-5 Ballute Jettisoning

Propulsion Systems. This group consists of the main propulsion system (MPS) and the reaction control system (RCS).

Main Propulsion System. The main engines are advanced LO_2/LH_2 expander cycle space engines, with retractable nozzles. The engines are rated at a maximum vacuum thrust of 5000 lbf each and provide a specific impulse of 483 sec. at an oxidizer-to-fuel mixture ratio of 6:1. Thrust vector control is provided by two electromechanical ball-screw linear actuators for each engine, equipped with redundant electric motor drives. Pressurization for the main propellant tanks is autogenous and consists of plumbing for delivery of pressurization gases (GH_2, GO_2) from the engine-mounted bleed ports to the tanks. Propellant feed lines are of aluminum and include bellows expansion joints to compensate for thermal expansion and engine gimballing. The propellant fill/drain/dump system includes rise-off disconnects for LH_2 and LO_2 at the OTV/ASE interface and provides for ascent abort dump of fully loaded main tanks (pressurant supplied from 3500 psia helium gas located on ASE). Two separate tank vent/relief systems are provided: one for use when stowed in the orbiter payload bay, and one for use in space. All main valves in the propellant feed system, and fill/drain/dump system are electrically actuated. All main propulsion systems including those in the ASE have dual failure tolerant redundancy to qualify as man-rated. Also included in the propellant subsystem are transfer lines for propellant feed and tank pressurization to the auxiliary tank set interface, for use when operating with an auxiliary tank set. These lines interface at the outer edge of the payload/tankset interface ring, with the LH_2 lines and LO_2 lines on opposite sides of the vehicle.

Reaction Control System. The RCS uses hydrazine monopropellant, pressurized by nitrogen gas supplied from a separate gas bottle. The thrusters (16 for unmanned configuration and 24 for a manned configuration) are located in four modules, and use a catalytic decomposition gas generator to produce 25 lb thrust each with a 320 psia supply pressure. Pressurant is supplied by a 3500 psia stored gas system, using a KEVLAR-overwrapped storage bottle. Propellant storage consists of 22 in. diameter titanium tanks with expulsion diaphragms (6 for delivery missions, and 8 for GEO sortie missions), each having a storage capacity of 195 lbs of hydrazine.

Thermal Protection and Control. Both active and passive techniques are used to provide thermal protection and control of the OTV.

Aeromaneuver Thermal Protection. Thermal protection for the ballute and vehicle during the aeromaneuver is provided by both flexible and rigid insulation. On the ballute heated face, and around the OTV body, flexible insulation is used, consisting of a NICALON cloth/Q-felt quilt. On the heat shield and engine nozzle doors, a rigid FRCI tile heat protection system is used, bonded to a strain isolation pad (SIP) on the graphite/polyimide (GR/PI) heat shield surface. For protection of the avionics and other sensitive equipment, a section of flexible quilted insulation is attached to the backside of the ballute to cover the equipment module during aeromaneuver.

Mission Thermal Control. Thermal control of the fuel cells is provided by an active thermal conditioning system consisting of a Freon 11 fluid loop with a radiator, located on the body shell exterior, and the associated pumps, valves, and control elements. The passive thermal control techniques include insulation blankets, thermal control coatings, and selected radiative surfaces. The thickness of the aluminum used for the avionics ring assembly is controlled to provide for proper heat flow from internally mounted components and its exterior surface is covered with flexible optical solar reflector (FOSR) to provide the radiative surface. Electrical heaters are provided for RCS components and avionics equipment as required. The LH_2 and LO_2 tanks are insulated using MLI. The MLI consists of layers of doubly aluminized kapton with a dacron net spacer. Thirty-four layers of MLI are used on the LH_2 tank and fifty layers on the LO_2 tank. The MLI wrapped tanks are enclosed within purge barriers which are purged with dry gas (helium for the LH_2 tank and nitrogen for the LO_2 tank) prior to launch.

Guidance and Navigation. The guidance and navigation subsystem consists of a laser gyro inertial reference unit (IRU), a star tracker, and a global positioning system (GPS) receiver-processor and antenna installation. The IRU and GPS are internally redundant. An additional star tracker is added for manned missions.

Communications and Data Handling. The communications subsystem consists of redundant radiofrequency (RF) links that are NASA STDN/TDRS compatible. Deployable pairs of antenna pods are diametrically located in the equipment ring assembly. Each RF link contains a 20W S-band power amplifier and a NASA STDN/TDRS transponder. For manned missions, the communications package is given additional redundancy by adding an extra transponder and amplifier.

The data handling subsystem consists of four advanced integrated data management units, each containing its own signal processing and conditioning units, and instrumentation for status monitoring of OTV subsystems. The data management units are sized to provide extra space for redundant data processing for manned missions. The instrumentation subsystem provides for monitoring of main propellant tank loading and usage.

Electrical Power. The primary power source is a set of lightweight-design O_2/H_2 fuel cells, each rated at 2 kW nominal, 3.5 kW peak, at a nominal 28V. The fuel cells are actively redundant (i.e., all operating in the normal mode) with each cell capable of providing normal mission power. The O_2/H_2 reactant is stored in the supercritical condition. Nominal operating pressures are 250 psia in the H_2 tank and 900 psia in the O_2 tank. The reactant tank set, which consists of a 24-in-diameter H_2 tank and a 19-in-diameter O_2 tank, has a storage capacity of 150 lb of usable reactant (energy capacity of approximately 180 kWh). Advanced Ni-H_2 utility batteries are provided for backup power and for smoothing of line transients. Power conversion and distribution are provided by redundant power distribution units, a power transfer unit, a pyro switching unit, and the vehicle cable harness. Not included in the vehicle cable harness are RF cabling within the communication subsystem and instrumentation wiring between sensors and data bus remote units.

Man-rating is provided by an extra active fuel cell, an extra utility battery, and an extra power transfer unit.

Airborne Support Equipment. The airborne support equipment (ASE) portion of the OTV main stage provide for the interfacing of the OTV to the Shuttle Orbiter. The ASE includes a structural subsystem, fluid subsystem, avionics, and electrical power. The main structure is fabricated from graphite/epoxy honeycomb sandwich and includes a shell, support rings, and longeron structures. Secondary structures include attach fittings and mechanisms, tilt mechanisms, and umbilical structures. The OTV interfaces with the ASE through a circular ring with a series of latching mechanisms and fluids/electrical umbilicals. On deployment, the combined OTV/ASE, is rotated to the release position, and the OTV is deployed by springs after release of the latching mechanisms. This process is shown in figure 2.3-6. Retrieval is performed in the reverse of this operation.

The ASE fluid system consists of orbiter-to-OTV fill, drain, dump, vent lines, umbilicals, and pressurant gas storage for abort dumping of propellant if required. These systems are dual-failure tolerant for man-rating. The abort dump pressurant gas is helium and is stored in KEVLAR-overwrapped high-pressure bottles.

The electrical and avionics subsystems provide backup electrical power and control, ASE status monitoring, and cabling and interfacing between OTV, spacecraft, and orbiter.

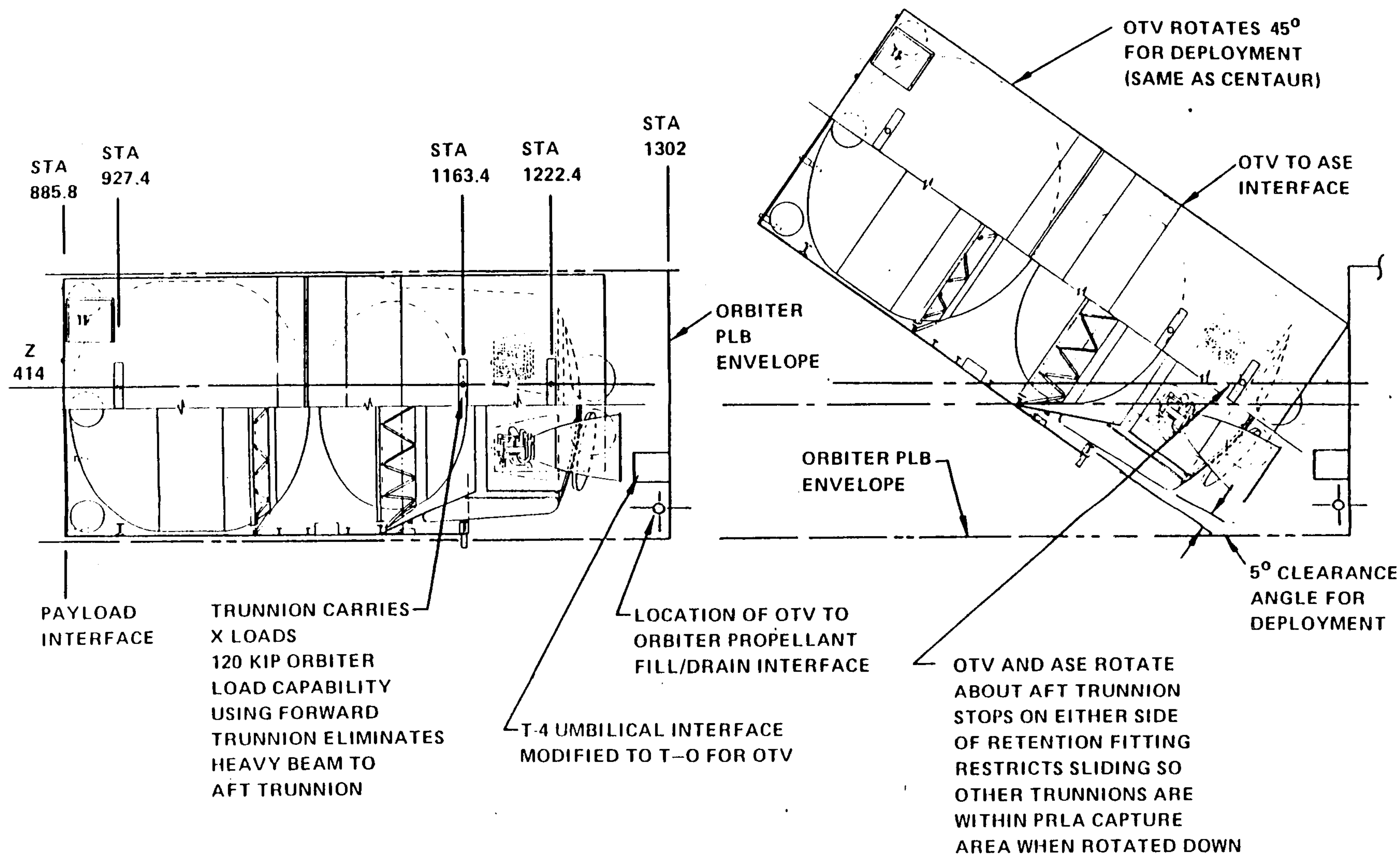

Figure 2.3-6 Shuttle Installation/Deployment – Ground Based Ballute Braked OTV

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2.4 AUXILIARY TANKSET

The configuration of the selected auxiliary tank set is shown in figure 2.4-1. The tankset is made up of the following subsystems:

- o Structural - includes an external, load-carrying shell, a LH_2 tank, and a LO_2 tank.
- o Propulsion systems - includes plumbing and tankset/OTV interfaces for main propellant feed, dump, pressurization, and venting.
- o Thermal control - includes passive thermal control of tanks.
- o Avionics/Electrical power - includes wiring and cabling harness and interfaces between payload and OTV.

The ASE is that portion of the auxiliary tank hardware system which remains in the Shuttle Orbiter bay when the auxiliary tanks are deployed. It provides for electrical, fluid, avionics, and structural interfaces between the auxiliary tankset and the orbiter. It also provides for abort dump pressurization.

A more detailed description of the auxiliary tankset subsystem is given in the following paragraphs.

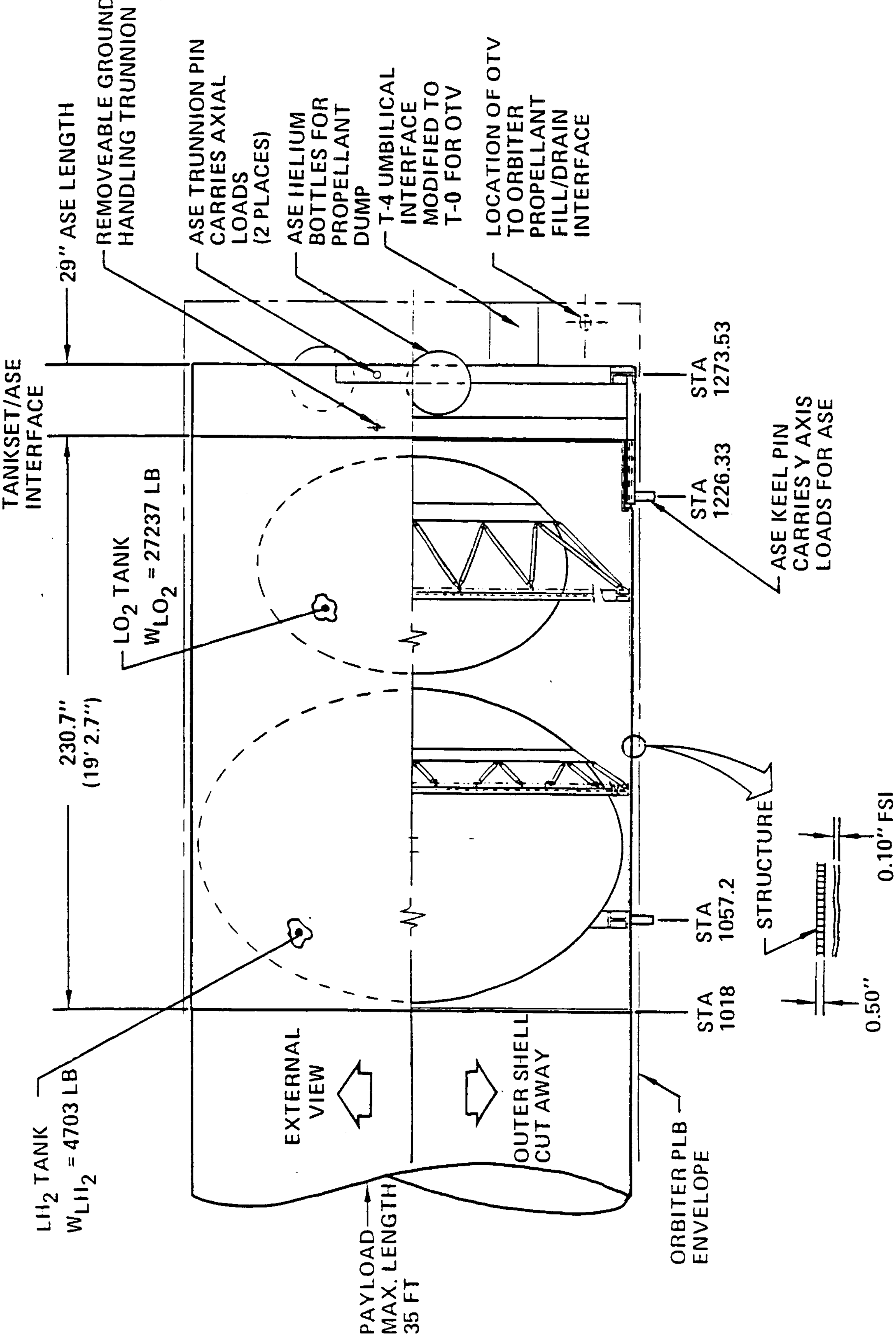
Structures. This group consists of the auxiliary tankset body structure and main propellant tankage.

Body Structure. The external shell supporting the auxiliary tankset tankage is similar in construction to that of the OTV main stage body shell. The shell is divided in 3 sections, with major rings at tank support locations. Secondary structures include payload support mechanisms, and orbiter handling and attachment fittings.

Main Propellant Tanks. The auxiliary tankset main propellant tanks have a design similar to those of the main stage, but are sized to a usable propellant capacity of 31,000 lbs at an oxidizer-to-fuel mixture ratio of 6:1.

Propulsion Systems. This group includes plumbing and interface fittings for propellant feed, fill, drain, and dump, as well as plumbing for tank pressurization and vent/relief. This plumbing is all similar to that on the OTV main stage and includes dual failure tolerant valve redundancy for man-rating.

Thermal Control. Thermal control of the propellant tanks is provided in a similar manner to that provided on the OTV main stage, with 34 layers of MLI on the LH_2 tank



OTV 905

Figure 2.4-1 Auxiliary Tankset Ground Based Ballute Braked OTV and 50 layers on the LO_2 tank. The MLI-wrapped tanks are enclosed within purge barriers which are purged with dry gas prior to launch. Also provided is a NICALON/ Q-felt flexible insulation quilt around the outside of the auxiliary tanks for aero-maneuver protection.

Avionics/Electrical Power. Avionics and electrical power included on the auxiliary tankset consists of wiring, cabling, and interfaces between the OTV main stage and the payload, as well as propellant usage monitors and instrumentation. No other independent power or data hardware is included on the tankset.

Airborne Support Equipment. The airborne support equipment portion of the auxiliary tankset provides the interfacing of the auxiliary tankset to the orbiter. The ASE includes a structural subsystem, fluid subsystem, avionics, and electrical power. The main structure is similar to that for the OTV main stage ASE, except that it is a shallower structure and does not include any tilt provisions. On deployment, the auxiliary tankset, with payload attached, is separated from the ASE and guided out of the payload bay, using the Shuttle Remote Manipulator System (RMS). The reason for the shorter ASE is to maximize the available payload volume.

The ASE fluid system and electrical/avionics subsystems are similar to those on the OTV main stage ASE, and perform the same functions.

2.5 WEIGHT SUMMARY

Weight summaries for the selected GB OTV concept involving main stage only and main stage plus auxiliary tankset for unmanned and manned missions are given in tables 2.5-1, 2.5-2, and 2.5-3, respectively. A weight growth margin of 5% on existing hardware and 15% on all new design was applied on all of the configurations. The two unmanned configurations include the structural/plumbing/cabling weight scar to accommodate additional redundancy associated with the manned missions.

The weight of the ballute aeroassist subsystem includes both structure and TPS, and is included under these subsystems in the preceding tables. The jettisonable portion of this subsystem is shown for each vehicle in table 2.5-4. The non-jettisonable portion, excluding weight growth allowance, is 438 lb of structure/inflation provisions, 93 lb of rigid TPS, and 293 lb of flexible body TPS. This is the same for all configurations.

Table 2.5-5 gives a weight summary of the ASE for both main stage and auxiliary tank set. The common GFE reflects equipment used by most payloads and involves a standard mix cabling harness (SMCH), CCTV, and utility kits. The standard allowance of 675 lbm for fittings has been removed however those fittings required specifically by the stage or auxiliary tank are incorporated as indicated.

Table 2.5-1. *Weight Summary Ground-Based, Ballute-Braked OTV – Multimanifest Mission*

- MAN RATED SUBSYSTEM SCAR
- 1500^{o} B/W, 1.5 T/D BALLUTE (33 FT. DIA.)
- SIZED FOR MULTI-MANIFEST MISSION

	LBS
STRUCTURES	3605
PROPULSION SYSTEMS	1251
THERMAL CONTROL AND PROTECTION	994
GUIDANCE, NAVIGATION, AND CONTROL	115
COMMUNICATIONS/DATA HANDLING	409
ELECTRICAL POWER	611
SPACE MAINTENANCE	N/A
WEIGHT GROWTH	976
(DRY WEIGHT)	(7961)
RESIDUALS	526
(INERT WEIGHT)	(8487)
REACTANTS–EPS (INC. RESERVES)	79
PROPELLANTS–RCS (INC. RESERVES)	624
MAIN PROPELLANTS (INC. RESERVES)	45981
FLUID LOSSES	338
(IGNITION WEIGHT)	(55509)
PAYLOAD	11000 UP/1000 DOWN
CONTRACTOR-FURNISHED ASE	3993
GOVERNMENT-FURNISHED ASE	2398
(LAUNCH WEIGHT)	(72900)
MASS FRACTION	.828

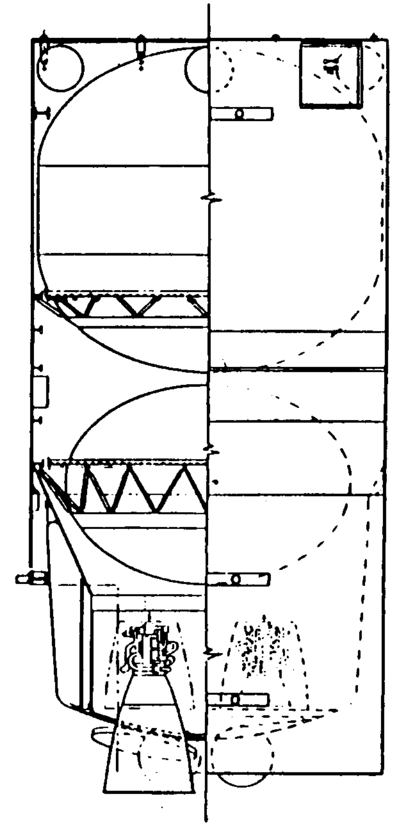


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Table 2.5-2. Weight Summary Ground-Based, Ballute-Braked OTV — 20k Delivery Mission

- Man rated subsystem SCAR, 1500° ballute (42 ft dia)
- Auxiliary tank sized for man sortie mission and offloaded for 20k low G delivery

	Main stage	Auxiliary tank + payload
Structures	3,786	1,730
Propulsion systems	1,326	410
Thermal protection and control	1,101	275
Guidance and navigation	115	–
Communications/data handling	409	75
Electrical power	611	110
Space maintenance	N/A	N/A
Weight growth	1,023	390
(Dry weight)	(8,371)	(2,990)
Residuals	526	230
(Inert weight)	(8,897)	(3,200)
Reactants – EPS (including reserves)	83	–
Propellants – RCS (including reserves)	1,411	–
Main propellants (including reserves)	45,981	25,782 (off-loaded)
Fluid losses	484	180
(Ignition weight)	(56,856)	(29,182)
Payload	–	20,000
Contractor-furnished ASE	3,933	2,792
Government-furnished ASE	2,397	1,809
(Launch weight, lbm) to S.S. orbit	(63,246)	(53,783)
Mass fraction	.809	.883

Table 2.5-3. Weight Summary Ground-Based, 1½ Stage, Ballute-Braked OTV — Manned Mission

- WEIGHT IN POUNDS
- MAN-RATED SUBSYSTEMS, ADDITIONAL RCS TANKAGE
- 1500° B/W, 1.5 T/D BALLUTE SIZED FOR 7.5K PAYLOAD RETURN (66 FT. DIA.)

	MAIN STAGE	AUX. TANKS
STRUCTURES	4494	1730
PROPULSION SYSTEMS	1348	410
THERMAL PROTECTION AND CONTROL	1521	275
GUIDANCE AND NAVIGATION	135	—
COMMUNICATIONS/DATA HANDLING	469	75
ELECTRICAL POWER	736	110
SPACE MAINTENANCE	N/A	N/A
WEIGHT GROWTH	1217	390
(DRY WEIGHT)	(9920)	(2990)
RESIDUALS	526	230
(INERT WEIGHT)	(10446)	(3220)
REACTANTS—EPS (INC. RESERVES)	37	—
PROPELLANTS—RCS (INC. RESERVES)	1563	—
MAIN PROPELLANTS (INC. RESERVES)	44489	30828
FLUID LOSSES	1832	180
(IGNITION WEIGHT)	(58367)	(34228)
PAYLOAD	—	7500
CONTRACTOR-FURNISHED ASE	3993	2792
GOVERNMENT-FURNISHED ASE	2398	1809
(LAUNCH WEIGHT) TO S.S. ORBIT	(64758)	(46329)
TOTAL MASS FRACTION	.813	.900

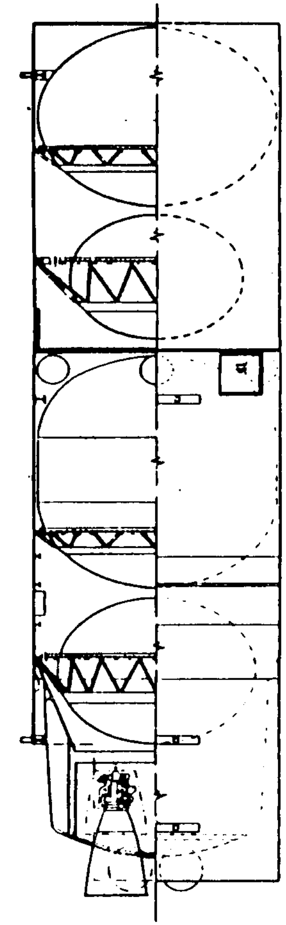

TABLE 2.5-4

GB BALLUTE WEIGHTS

Item	UMM Delivery	20k Delivery	7.5k Roundtrip
Diameter	33 ft.	42 ft.	66 ft.
Ballute Structure-Jettisonable	(545)	(725)	(1404)
Fabric Structure	316	479	1100
Support Structure	229	246	304
Thermal Protection-Jettisonable	(264)	(373)	(796)
Flexible	248	357	780
Rigid	16	16	16
Weight Growth	(121)	(165)	(330)
TOTAL JETTISONABLE WEIGHT	930	1263	2530

Table 2.5-5 Airborne Support Equipment (ASE) for Ground-Based SCB OTV System

WEIGHT AT LAUNCH (LBM)

	PAYLOAD ONLY	PAYLOAD & AUX. TANK	PAYLOAD & STAGE	STAGE ONLY
SCB EQUIPMENT				
COMMON GFE	323	323	323	323
FITTING ALLOWANCE	-675	-675	-675	-675
STD PAYLOAD EQUIPMENT	998	998	998	998
PAYLOAD SUPPORT EQUIPMENT	936	100	100	—
ATTACH FTGS.	836			—
P/L KITS	100	100	100	—
CONTRACTOR - FURN. STRUCTURE	— *	— *	— *	—
AUX. TANK EQUIPMENT	—	4161	—	—
ATTACH FTGS.	—	1086	—	—
FLUID SYSTEMS, ETC.	—	285	—	—
CONTRACTOR - FURN. STRUCTURE	—	2790	* —	—
OTV RETURN EQUIPMENT	—	—	6068	6068
ATTACH FTGS.	—	—	1705	1705
PRESS. SYSTEMS, TOOLS	—	—	370	370
CONTRACTOR - FURN. STRUCTURE	—	—	3993	3993
TOTAL	1259	4584	6491	6391

* INCL. IN PAYLOAD WEIGHT

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2.6 C. G. MASS CHARACTERISTICS

The longitudinal center of gravity compatibility of the OTV mainstage/payload and auxiliary tankset/payload with Orbiter requirements is presented in figures 2.6-1 and 2.6-2. As shown, Orbiter requirements are met with the exception of the full auxiliary-tankset-only launch. However, manifesting such a launch with another payload would draw the c.g. forward within the bounds.

2.7 RELIABILITY/REDUNDANCY/MAN-RATING

The vehicle reliability target for unmanned missions is 0.995 based on cost optimization analysis. Figure 2.7-1 displays the means used to determine the cost optimum point as well as a few of the more significant components requiring redundancy. Most notable of these components are the addition of a second main engine and fuel cell, additional data management units and RCS thrusters. The cost optimum point reflects the situation of development, production, and launch cost going up with higher reliability levels beyond single thread but reflight (because of failure) going down significantly. To achieve this point, a total of 45 additional components were added to the single thread configuration resulting in $2M savings per flight relative to a configuration with a complete set of redundant components.

For manned missions (only 3 beginning in 2008), the vehicle is configured so there is no credible single point failures. In essence this means dual failure tolerant. To achieve this configuration a component weight scar of 418 lbm results.

2.8 PERFORMANCE

The selected concept OTV is a ground based vehicle using an expendable ballute for aerobraking. Its main propulsion system uses LO2/LH2, and consists of 2 advanced engines with an expansion ratio of 1000, thrust of 5000 lbs per engine, and specific impulse of 483.2 seconds. Its auxiliary propulsion system uses Hydrazine, and has a specific impulse of 220 seconds. The assumptions used for all performance analyses are detailed in Volume II, Book 3, Section 9.

2.8.1 Vehicle Sizing

The selected concept OTV is designed to satisfy the low mission model, with three missions acting to size the vehicle's propellant capacity. These missions, in the order as they appear in the mission model are:

1. 10,000 lb (net) GEO unmanned multiple manifest (11 k lb up/1 k lb down)
2. 20,000 lb GEO unmanned low g
3. 7500 lb GEO manned sortie

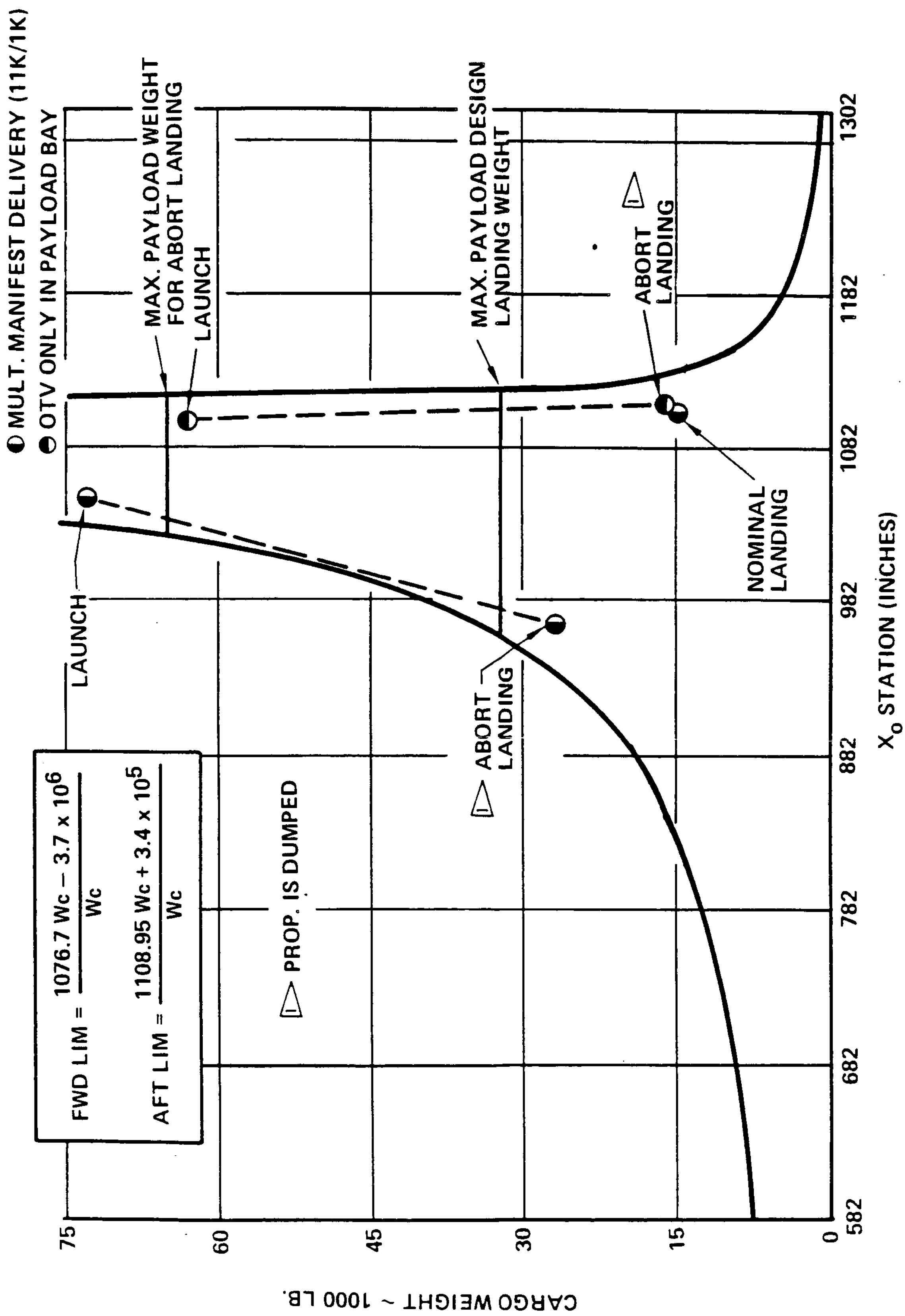

Figure 2.6-1 GB Ballute-Braked OTV Launch, Orbiter Cargo C.G. Limits
Ref. — JSC 07700 Vol. XIV

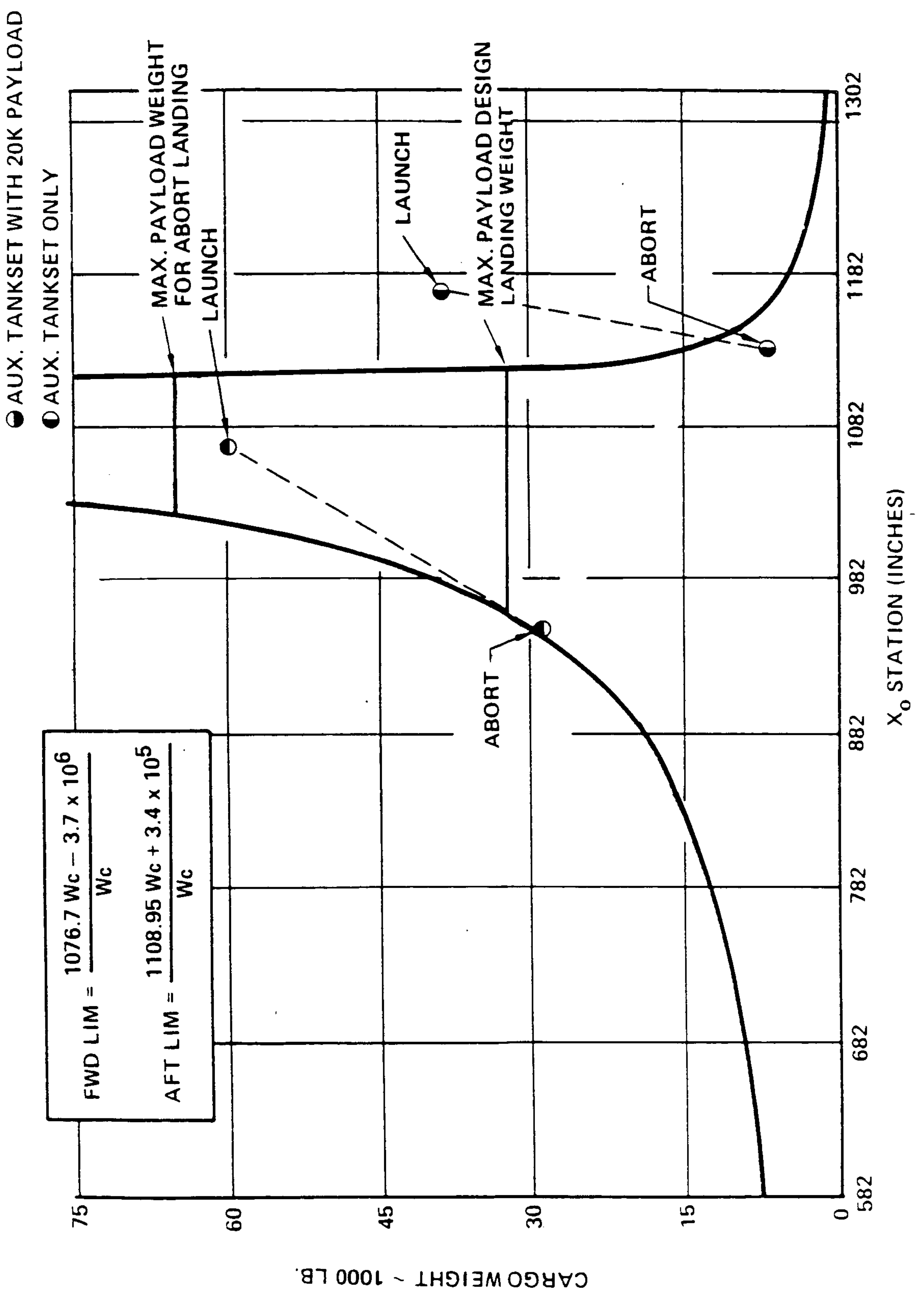

Figure 2.6-2 GB Auxiliary Tank Launch, Orbiter Cargo C.G. Limits
Ref. — JSC 07700 Vol. XIV

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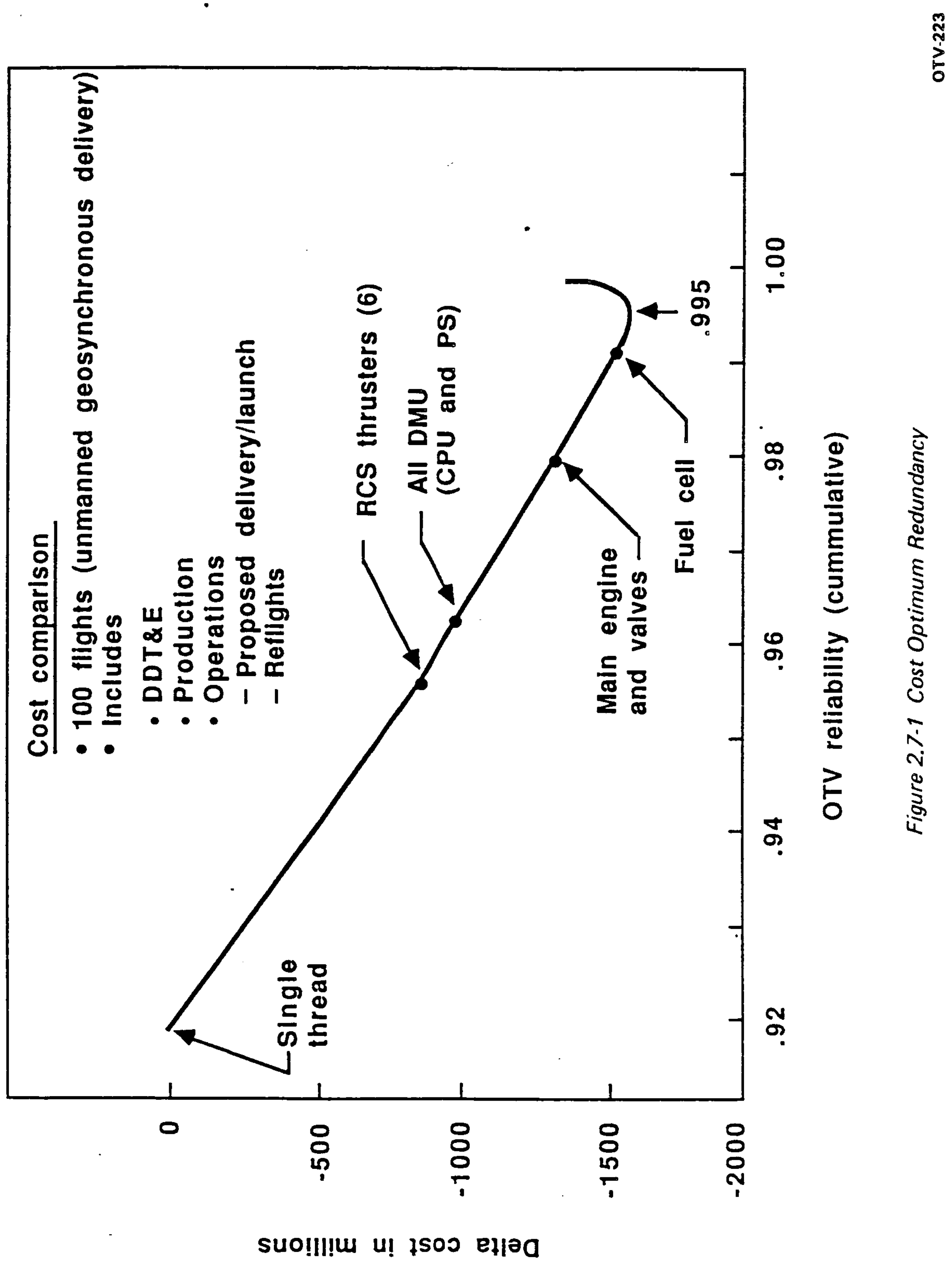

Figure 2.7-1 Cost Optimum Redundancy

The concept involves the design of a single main stage, 3 sizes of expendable ballutes, and one size of integral auxiliary propellant tanks. The main stage initially is not man-rated, and is modified when required. The 3 ballutes are sized based on the atmospheric reentry weights, which are significantly different for three missions. The integral auxiliary tanks are mated with the main stage at the space station, and the additional propellant enables the vehicle to perform the more demanding missions. The auxiliary tanks are sized for the manned sortie mission, which requires the most MPS propellant for this concept.

A major OTV sizing constraint is that of the Shuttle's payload capability. The capability to a 120 nm circular orbit for 2 different inclinations of interest was taken to be:

Inclination	Shuttles Payload Capability
28.5 degrees	74280 lbs
57.0	57300

This capability per study groundrule assumes Orbiter No. 104 dry weight, filament wound case SRB's, and SSME's operating at 109%.

The following subsections detail the vehicle sizing for the three missions described above. A summary of these sizings is given in table 2.8.1-1

GEO Unmanned Multiple Manifest Mission. The 10,000 lb (net) GEO unmanned multiple manifest mission is taken to be the mission which sizes the main propellant tanks and 33 ft dia ballute aerobrake. This mission's sequence of events begins with the OTV being deployed from the shuttle into a 120 nm circular orbit. The OTV then proceeds to GEO, deploys the payloads, deorbits, aerobrakes, and circularizes into a 150 nm circular orbit, awaiting pickup by the shuttle. The payload rack which is carried throughout the mission, weighs 1000 lbs. The computer sizing output is shown in table 2.8.1-2. Note that the initial weight of the OTV and payload is 66258 lbs, quite close to the shuttle's maximum usable capability of 67889 lbs.

GEO Unmanned Low G Mission. The 20,000 lb GEO unmanned low g mission cannot be performed by the vehicle sized for the multiple manifest mission. Using a vehicle with larger sized propellant tanks would be beyond the payload capacity of our assumed shuttle. Therefore, for this mission, the payload and an auxiliary tank set are mated with the main stage at the Space Station or STS Orbiter. The OTV then transits to GEO, deposits it's payload, deorbits, aerobrakes, and circularizes into a 150 nm circular orbit. The auxiliary tanks are sized for the propellant requirements of the most demanding

Table 2.8.1-1. Selected Concept Vehicle Sizing Summary

Sizing Mission	Multi-Manifest	Low G	Sortie
Total MPS propellant (lbs)	46800	72948	77881
Usable MPS propellant			
Core stage	45981	45981	44489
Auxiliary tanks	—	25782	30828
End of mission weight			
Core stage	8546	8631	8806
Auxiliary tanks	—	3848	3848
Jettisonable ballute	930	1263	2530

Table 2.8.1-2 10000 lb GEO Unmanned Multiple Manifest Performance Summary

7:12 AM, 2-AUG-85

BASING/MISSION: GROUND BASED GEO UNMANNED MULTIPLE MANIFEST
BRAKE: EXPENDABLE BALLUTE, B/W TEMP = 1500, T/D = 1.5
ENGINE: 2 ADVANCED, THRUST = 10000.(TOTAL)
PROPULSION: MPS ISP = 483.2, ACS ISP = 220.0
MAIN TANK SIZING: GROUND BASED GEO UNMANNED MULTIPLE MANIFEST
BRAKE SIZING: GROUND BASED GEO UNMANNED MULTIPLE MANIFEST
STAGES: 1

WEIGHTS INPUT (LBS)

STAGE END = 8546., JETT BALLUTE = 930.
STAGE TRENDING: END OF MISSION = 5588. + .0644 * MPS USABLE
BALLUTE TRENDING: JETTISON = 922. + .0002 * MPS USABLE

MISSION PROFILE	DELTA V (F/S)	DELTA T (HOURS)	DELTA W (LBS)	WEIGHT (LBS)
1 ACS COAST IN 120 NM ORBIT	10.	0.8	-98.	66430.
2 MPS PERIGEE BURN 1	3649.	0.2	-14031.	52399.
3 ACS COAST	20.	3.0	-163.	52236.
4 MPS PERIGEE BURN 2	4459.	0.2	-13135.	39101.
5 ACS COAST	20.	5.3	-138.	38963.
6 MPS APOGEE BURN	5865.	0.1	-12262.	26702.
7 ACS PAYLOAD POSITIONING	15.	1.0	-62.	26640.
8 DROP PAYLOAD	0.	0.0	-10000.	16640.
9 ACS COAST	50.	24.0	-241.	16399.
10 MPS DEORBIT BURN	6245.	0.1	-5442.	10957.
11 ACS COAST	10.	6.2	-48.	10909.
12 MPS BURN	50.	0.1	-60.	10850.
13 AEROMANEUVER	0.	0.1	-930.	9919.
14 MPS POST AERO CORRECT	225.	0.1	-167.	9752.
15 ACS COAST	10.	0.8	-18.	9734.
16 MPS BURN	216.	0.1	-159.	9575.
17 ACS COAST	10.	3.0	-29.	9546.
18 DROP RACK	0.	0.0	-1000.	8546.

GRAVITY/STEERING LOSS (F/S) = 70.

PROPELLANT SUMMARY (LBS)

MPS TOTAL PROPELLANT = 46800.

MPS USABLE =	45981.	ACS USABLE =	624.	EPS USABLE =	79.
NOMINAL =	45080.	NOMINAL =	567.	NOMINAL =	66.
RESERVES =	902.	RESERVES =	57.	RESERVES =	13.
BOILOFF =	163.				
START/STOP =	175.				

mission, the manned sortie mission. The OTV with auxiliary tank is 2800 lbs heavier than the vehicle for the multiple manifest mission at the time of aeromaneuver and thus the ballute diameter increases to 40 ft. The low g mission requires throttling of the main engines to keep the acceleration at or below 0.1 g's until payload deployment. The resulting thrust level reduces the nominal Isp by 2 seconds. The performance summary for this mission is shown in table 2.8.1-3

GEO Manned Sortie Mission. The 7500 lb manned GEO sortie mission requires the most MPS propellant. Like the low g mission, the OTV main stage is mated with its auxiliary tanks and payload at the space station before travelling to GEO. However, the main stage is modified to be man-rated and the ballute must be 65 ft diameter to allow for the heavier aeromaneuver weight with the manned cab. The auxiliary propellant tanks are sized for the propellant requirements of this mission. The computer sizing output for this mission is shown in table 2.8.1-4.

2.8.2 Vehicle Performance/Mission Propellant

The performance of the three vehicle configurations that make up the selected concept that satisfies the various missions are shown in table 2.8.2-1. The propellant requirements as a function of payload weight (in lbs) are given for all missions in the form of propellant coefficients:

MPS Total Propellant = A + (B x Payload Weight)

The maximum payload deliverable for each configuration and mission are given as well, since for a few cases this maximum occurs not because of insufficient MPS propellant tank capacity, but because it is beyond the payload capacity of our assumed shuttle.

2.9 LAUNCH AND RECOVERY OPERATIONS

The launch and recovery operations associated with each major mission class and the corresponding users charge load factor is presented in table 2.9-1. A full load factor (1.0) occurs when either the mass or length of cargo reaches three quarters of the capacity. GEO delivery missions (Class 1) involving less than 12K lbm equivalent involve only the use of the main stage and a single STS flight. Class 2 DOD 12K lbm missions also only involve the main stage but due to payload length a second STS is required. Class 3, 4, and 5 missions all require use of the OTV main stage and auxiliary propellant tank. Due to mass as well as length, two STS flights are required to support these missions.

Table 2.8.1-3 20000 lb GEO Unmanned Low G Performance Summary

7:43 AM, 2-AUG-85

OTV MPS USABLE PROPELLANT FIXED AT 45981LBS
BASING/MISSION: GROUND BASED GEO UNMANNED LOW G
BRAKE: EXPENDABLE BALLUTE, B/W TEMP = 1500, T/D = 1.5
ENGINE: 2 ADVANCED, THROTTLED TO G LIMIT = 0.10
PROPULSION: MPS ISP = 481.2, ACS ISP = 220.0
MAIN TANK SIZING: GROUND BASED GEO UNMANNED MULTIPLE MANIFEST
BRAKE SIZING: GROUND BASED GEO UNMANNED LOW G
AUX TANK SIZING: GROUND BASED GEO MANNED SORTIE
STAGES: 1 WITH REUSABLE AUXILLIARY TANKS

WEIGHTS INPUT (LBS)

STAGE END = 8631., JETT BALLUTE = 1263., AUX TANK END = 3848.

MISSION PROFILE	DELTA V (F/S)	DELTA T (HOURS)	DELTA W (LBS)	WEIGHT (LBS)
1 MPS BURN FROM 120 NM. CIRC	261.	0.1	-974.	55810.
2 ACS COAST	10.	0.8	-83.	55727.
3 MPS BURN TO 270 NM. CIRC	259.	0.1	-949.	54778.
4 ACS REND/DOCK	40.	1.0	-314.	54464.
5 DOCKED AT LEO STATION	0.	24.0	-89.	54375.
6 ATTACH AUX TANK	0.	0.0	29149.	83525.
7 PICKUP PAYLOAD	0.	0.0	20000.	103525.
8 ACS SEPARATION	10.	0.0	-146.	103378.
9 ACS COAST	10.	0.8	-150.	103228.
10 AUX TANK PERIGEE BURN 1	4217.	0.2	-25301.	77927.
11 ACS COAST	20.	3.0	-235.	77692.
12 MPS PERIGEE BURN 2	3639.	0.2	-16633.	61059.
13 ACS COAST	10.	5.3	-114.	60945.
14 MPS BURN	5798.	0.1	-19054.	41891.
15 ACS PAYLOAD POSITIONING	15.	1.0	-94.	41797.
16 DROP PAYLOAD	0.	0.0	-20000.	21797.
17 ACS COAST	50.	24.0	-277.	21520.
18 MPS DEORBIT BURN	6245.	0.1	-7160.	14360.
19 ACS COAST	10.	6.2	-52.	14307.
20 MPS BURN	50.	0.1	-71.	14236.
21 AEROMANEUVER	0.	0.1	-1263.	12973.
22 MPS POST AERO CORRECT	257.	0.1	-238.	12735.
23 ACS COAST	10.	0.8	-22.	12713.
24 MPS BURN	216.	0.1	-201.	12512.
25 ACS COAST	10.	3.0	-33.	12479.
26 DETACH AUX TANK	0.	0.0	-3848.	8631.

GRAVITY/STEERING LOSS (F/S) = 218.

PROPELLANT SUMMARY (LBS)

MPS TOTAL PROPELLANT = 72948.

MPS USABLE =	45981.	ACS USABLE =	1411.	EPS USABLE =	83.
NOMINAL =	45080.	NOMINAL =	1283.	NOMINAL =	69.
RESERVES =	902.	RESERVES =	128.	RESERVES =	14.
BOILOFF =	259.				
START/STOP =	225.				

AUXILLIARY TANKS

USABLE =	25782.
NOMINAL =	25276.
RESERVES =	506.

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Table 2.8.1-4 7500 LB GEO Manned Sortie Summary

7:31 AM, 2-AUG-85

OTV MPS USABLE PROPELLANT FIXED AT 44489 LBS
BASING/MISSION: GROUND BASED GEO MANNED SORTIE
BRAKE: EXPENDABLE BALLUTE, B/W TEMP = 1500, T/D = 1.5
ENGINE: 2 ADVANCED, THRUST = 10000.
PROPULSION: MPS ISP = 483.2, ACS ISP = 220.0, MGS ISP = 220.0
MAIN TANK SIZING: GROUND BASED GEO UNMANNED MULTIPLE MANIFEST
BRAKE SIZING: GROUND BASED GEO MANNED SORTIE
STAGES: 1 WITH REUSABLE AUXILLIARY TANKS

WEIGHTS INPUT (LBS)

STAGE END = 8806., JETT BALLUTE = 2530., AUX TANK END = 3848.
STAGE TRENDING: END OF MISSION = 5947. + .0643 * MPS USABLE
BALLUTE TRENDING: JETTISON = 1975. + .0180 * AUX TANK USABLE
AUX TANK TRENDING: END OF MISSION = 1744. + .0683 * AUX TANK USABLE

MISSION PROFILE	DELTA V (F/S)	DELTA T (HOURS)	DELTA W (LBS)	WEIGHT (LBS)
1 MPS BURN FROM 120 NM. CIRC	261.	0.1	-994.	57217.
2 ACS COAST	10.	0.8	-85.	57132.
3 MPS BURN TO 270 NM. CIRC	259.	0.1	-969.	56164.
4 ACS REND/DOCK	40.	1.0	-322.	55842.
5 DOCKED AT LEO STATION	0.	24.0	-89.	55753.
6 ATTACH AUX TANK	0.	0.0	34097.	89851.
7 PICKUP PAYLOAD	0.	0.0	7500.	97351.
8 ACS SEPARATION	10.	0.0	-137.	97213.
9 ACS COAST	10.	0.8	-141.	97072.
10 AUX TANK PERIGEE BURN 1	5600.	0.2	-30249.	66823.
11 ACS COAST	20.	3.0	-204.	66619.
12 MPS PERIGEE BURN 2	2256.	0.2	-9048.	57571.
13 ACS COAST	10.	5.3	-109.	57462.
14 MPS BURN	5798.	0.1	-17905.	39557.
15 ACS REND/DOCK	40.	1.0	-228.	39329.
16 ATTACH MGS	0.	0.0	38374.	77703.
17 MGS OPERATIONS	20.	24.0	-308.	77395.
18 DROP MGS PAYLOAD	0.	0.0	-426.	76969.
19 MPS BURN	46.	0.1	-252.	76716.
20 MGS COAST	10.	48.5	-288.	76429.
21 MPS BURN	46.	0.1	-251.	76178.
22 MGS OPERATIONS	60.	24.0	-731.	75447.
23 DROP MGS PAYLOAD	0.	0.0	-427.	75020.
24 MPS BURN	160.	0.1	-793.	74227.
25 MGS COAST	10.	126.0	-570.	73657.
26 MPS BURN	160.	0.1	-779.	72878.
27 MGS OPERATIONS	60.	24.0	-703.	72175.
28 DROP MGS PAYLOAD	0.	0.0	-429.	71746.
29 MPS BURN	46.	0.1	-237.	71509.
30 MGS COAST	10.	48.5	-280.	71229.
31 MPS BURN	46.	0.1	-235.	70993.
32 MGS OPERATIONS	70.	48.0	-874.	70119.
33 DROP MGS PAYLOAD	0.	0.0	-427.	69692.
34 MGS COAST	10.	6.0	-121.	69571.
35 DETACH MGS	0.	0.0	-34081.	35490.
36 ACS SEPARATION	10.	0.0	-50.	35440.
37 MPS DEORBIT BURN	6245.	0.1	-11741.	23699.
38 ACS COAST	10.	6.2	-66.	23634.
39 MPS BURN	50.	0.1	-101.	23533.
40 AEROMANEUVER	0.	0.1	-2530.	21003.
41 MPS POST AERO CORRECT	42.	0.1	-82.	20922.
42 ACS COAST	10.	0.8	-34.	20888.
43 MPS BURN	420.	0.1	-581.	20307.
44 ACS COAST	10.	0.8	-33.	20274.
45 ACS REND/DOCK	40.	1.0	-119.	20155.
46 DROP PAYLOAD	0.	0.0	-7500.	12655.
47 DETACH AUX TANK	0.	0.0	-3848.	8806.

GRAVITY/STEERING LOSS (F/S) = 209.

PROPELLANT SUMMARY (LBS)

MPS TOTAL PROPELLANT = 77881.

MPS USABLE	= 44489.	ACS USABLE	= 1563.	EPS USABLE	= 37.
NOMINAL	= 43617.	NOMINAL	= 1420.	NOMINAL	= 31.
RESERVES	= 872.	RESERVES	= 142.	RESERVES	= 6.
BOILOFF	= 1457.				
START/STOP	= 375.				

AUXILLIARY TANKS
USABLE = 30828.
NOMINAL = 30224.
RESERVES = 604.

MGS NOMINAL = 2584.

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Table 2.8.2-1 Selected Concept Vehicle Performance

Mission	Propellant coefficients A	Propellant coefficients B	Maximum Payload (lbs)
Configuration Sized For The 10000 lb GEO Unmanned Multiple Manifest Mission:			
GEO Unmanned	28447	1.544	11924
GEO Unmanned Multiple Manifest	31365	1.548	10000
GEO Unmanned Low G	29292	1.553	11258
Lunar Unmanned	23467	1.425	14511 (Shuttle Constrained)
Planetary, C3 = 28 Km^2/sec^2	25484	1.988	11126
Molniya	11725	0.888	15917 (Shuttle Constrained)
Configuration Sized For The 20000 lb GEO Unmanned Mission:			
GEO Unmanned	41223	1.531	23943
GEO Unmanned Multiple Manifest	44177	1.535	21957
GEO Unmanned Low G	42257	1.529	23299
Lunar Unmanned	32435	1.438	31603
Planetary, C3 = 28 Km^2/sec^2	38997	2.122	18324
Configuration Sized For The 7500 lb GEO Manned Sortie Mission:			
GEO Manned Sortie Mission	54396	3.243	7500

OTV 1715

Table 2.9-1 GB SCB OTV Launch and Recovery Operations

● 72K STS

MISSION CLASS	NUMBER FLIGHTS (GEO/HI INCL)	FLIGHT OPERATIONS	USERS CHARGE LOAD FACTOR [1] GEO	USERS CHARGE LOAD FACTOR [1] HI INCL
(1) GEO 12/0 AND 11/1 KLBS ≤ 25FT	65	● STS 1 – MAIN STAGE + RASE + PAYLOAD TO DEPLOYMENT ORBIT ● OTV RETURNS TO STS 1	1.0 (W)	–
(2) DOD 12/O KLBS 30 FT	28 (17/11)	● STS 2 – MAIN STAGE + RASE ● STS 1 – PAYLOAD + ASE ● OTV RETURNS TO STS 2	1.0(W) 0.73(L)	0.69(L) 0.73(L)
(3) DOD 20 KLBS 35 FT	40 (24/16)	● STS 2 – MAIN STAGE + RASE ● STS 1 – AUX TANK + PAYLOAD + ASE ● OTV RETURNS TO STS 2	1.0(W) 1.0(W)	0.76(W) 0.77(L)
(4) GEO 20 KLBS 35 FT	9	● STS 1 – AUX TANK + PAYLOAD + ASE ● STS 2 – MAIN STAGE + RASE ● OTV ELEMENTS INTEGRATED AT STATION OTV RETURNS TO STS 2	1.0(W) 1.0(W)	-- --
(5) GEO MAN SORTIE 7.5K/10 FT	3	● STS 1 – AUX TANK + PAYLOAD + ASE ● STS 2 – MAIN STAGE + RASE ● OTV ELEMENTS INTEGRATED AT STATION	1.0(W) 0.77(W)	– –
	145			

RASE = RETURN AIRBORNE SUPPORT EQUIP

[1] COST CRITERIA: (W) WEIGHT; (L) LENGTH ;1.0 = FULL CHARGE =$73M

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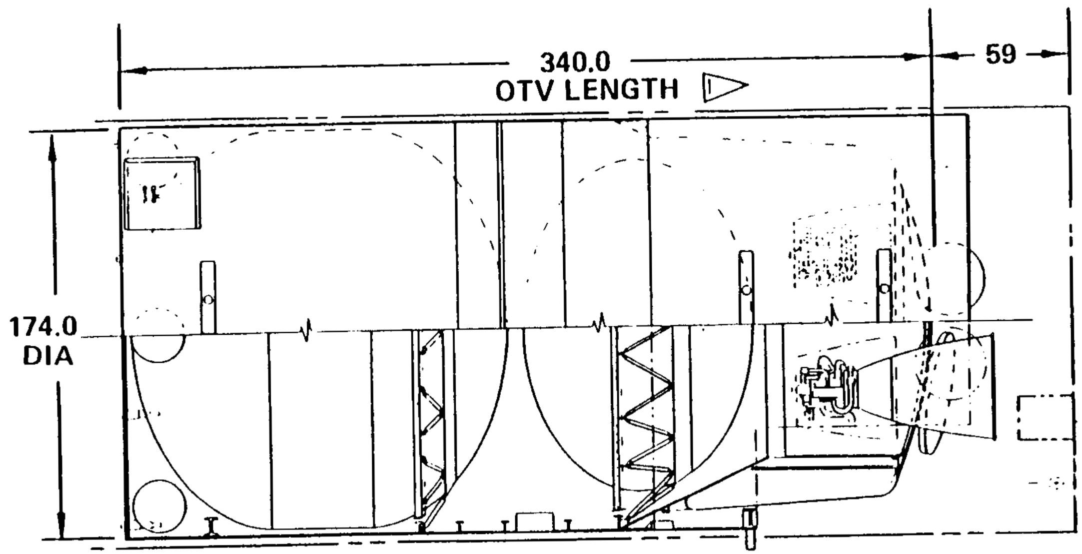
2.10 IMPACT OF 65K STS

Part of the focus of the fifth quarter work was to address the impact on the OTV system of having Shuttle lift capability of 65,000 lb, rather than the 72,000 lb set as an original groundrule. This caused a decrease in main stage propellant loading, with a corresponding decrease in payload delivery capability and stage length. The main stage configuration and summary weight is shown in figure 2.10-1. This stage has a propellant load capacity of 40,850 lb and will delivery 9000 lb to GEO. In order to perform the other missions required in the mission model, auxiliary tanksets must be added to the main stage. To perform the 12K GEO delivery mission, a small tankset with 15,000 lb propellant capacity must be used. For the 20K GEO delivery, and 7.5K manned sortie missions, a large tankset, having 39,600 lb propellant capacity, must be used. This tankset is shown in figure 2.10-2. Weight summaries for these various missions are shown in table 2.10-1. Launch weights shown over 65,000 lb reflect the necessity to make two STS launches to complete the mission.



FEATURES

- LAUNCHED AND RETURNED IN ORBITER CARGO BAY
- BALLUTE AEROASSIST—EXPENDABLE
  - –33 FT. DIA FOR MAIN STAGE RETURN
  - –36 FT. DIA FOR RETURN W/SMALL AUX TANKS
  - –43 FT. DIA FOR RETURN W/LG. AUX TANKS
  - –67 FT. DIA FOR RETURN OF AUX. TANKS & CAB
- ADVANCED CRYO ENGINES T = 5000 LBF EACH
- 2 TANKS

WEIGHT SUMMARY (LBM)

	MAN-SCAR (33 FT. BALLUTE)	MAN-RATED (67 FT. BALLUTE)
• DRY	7,784	9,684
• MAIN PROP (TOTAL)	40,850	40,850
• OTHER FLUIDS	1,173	1,242
• START BURN	49,807	51,776

Figure 2.10-1 GB SCB OTV Configuration Update – 65K Orbiter Launch

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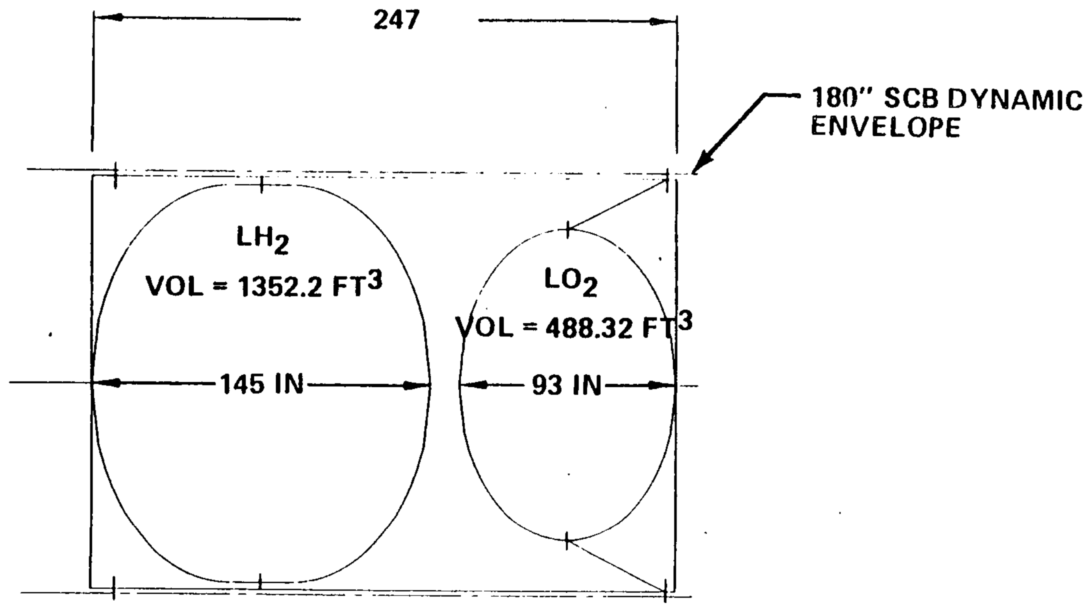

FEATURES

- ONE LH_2 TANK 0.707 ELLIPTICAL HEADS
- ONE LO_2 TANK 0.707 ELLIPTICAL HEADS
- OCB-- CARRIED THRU 4 TRUNNIONS AND ONE KEEL PIN

WEIGHT SUMMARY (LBM)

• USABLE PROPELLANT,	39,600
• INERT	3,673
• TOTAL	43,273

Figure 2.10-2 39.6k Auxiliary Tank – GB SCB OTV

OTV-1153

Table 2.10-1 GB SCB OTV Weight Summary

MAIN STAGE SIZED FOR 65K STS

	9K GEO DELIV. [1]	12K GEO DELIV.	20K GEO DELIV.	7.5K MANNED RETURN
STAGE DRY WEIGHT	7,784	7,886	8,151	9,684
AUX. TANK DRY WEIGHT	---	2,449	3,286	3,286
TOTAL MPS PROPELLANT	40,850	55,620	73,840	82,590
OTHER FLUIDS	975	1,345	1,542	1,804
PAYLOAD	9,000	12,000	20,000	7,500
START-BURN WEIGHT	58,609	79,300	106,819	104,864
ASE	6,391	[2] 10,975	[2] 10,975	[2] 10,975
ACC	--	--	--	--
TOTAL LAUNCH WEIGHT	65,000	90,275	117,794	115,839

[1] 65K STS LIMIT SIZING

[2] REFLECTS ASE FOR TWO STS LAUNCHES (1ST–P/L + AUX. TANK; 2ND–STAGE)

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3.0 SUBSYSTEMS DESCRIPTION

Each subsystem of the vehicle is discussed in terms of its key requirements, hardware and functional description and detail weight breakdown.

3.1 STRUCTURE

The current GB Ballute Brake OTV is an outgrowth of the vehicle defined in "Orbital Transfer Vehicle Concept Definition Study" Final Report Volume 4 "Selected Concept Definition" D180-26090-4, 1980 under NASA contract NAS8-33532. Basic structural concept selections presented in that report and retained in the current configuration have been reevaluated as the configuration has evolved. However, those selections, in general, are not rejustified by detailed trades in this report. Key results of structural analyses conducted to support the structural design/evaluations are presented in Section 3.2 of Volume II, Book 3.

A key distinction of the GB OTV compared to the SB OTV is that it is launched fully loaded in the Orbiter and is sized to take maximum advantage of the Orbiters launch capability. Therefore most of the structure in the GB OTV is designed for the Orbiter induced launch loads and the OTV free flight loads are quite small by comparison.

3.1.1 Structural Requirements

Structural design requirements for the OTV studies have been taken from applicable sections from References 3-1 through 3-6.

a. Reference 3-1, MSFC-HDBK-505, "Structural Strength Program Requirements".
b. Reference 3-2, VSC 07700 Volume XIV Attachment 1 (ICD 2-19001), "Shuttle Orbiter/Cargo Standard Interfaces".
c. Reference 3-3, D290-10050-1, "IUS Structural Design Criteria".
d. Reference 3-4, NASA TM 82585, "Natural Environment Design Criteria".
e. Reference 3-5, NASA TM 82478, "Space and Planetary Environment Criteria Guidelines for Use in Space Vehicle Development".
f. Reference 3-6, VSC-20001, "Orbital Debris Environment for Space Station".

References 3-1, -2 and -3 were used as a composite structural design criteria for all OTV structural systems. Reference 3-1 was used for all space operations remote from the Orbiter. Reference 3-2 was used to define interfaces and load factors while in the Orbiter and 3-3 was used for the remaining structural requirements applicable to operation in the Orbiter.

References 3-4, -5 and-6 were used to define meteoroid and space debris environments at the space station and at GEO.

A summary of the key structural design criteria/guidelines is presented in Table 3.1.1-1.

The Orbiter lift-off loads were derived from reference 3-2, and reflect the Shuttle cargo bay conditions. The factor of safety of 1.5 was selected for consistency with that used in earlier OTV vehicle design studies to permit incorporation and extrapolation of that design data. A factor of 1.4 is commonly used to design for loads experienced while in the Orbiter and lessor load factors are frequently used for space systems remote from the Orbiter. However those lower load factors are commonly used for single launch systems in which fatique is not a required consideration. The use of the 1.5 factor at this point in these design studies actually provides an allowance for fatigue which can be properly accounted for along with a correspondingly lower factor of safety at the time of detailed design of the structure.

For tankage design the 1.5 factor of safety on yield is satisfied but it does not control the design. The tankage design stress level is established by the room temperature proof test required to verify the applicable fatigue life.

For meteoroid/debris shielding, the reliability requirement is set by overall mission reliability requirements. This requirement is to be met, given the environments shown in figure 3.1.1-1.

3.1.2 Vehicle Structural Definition

The major vehicle structural elements of the Ground Based, Ballute-braked OTV are as follows:

a. Aerobrake,
b. Thrust Structure,
c. Equipment Support Section,
d. Propellant Tanks,
e. Body Structure,
f. Rings Integral with Tanks,
g. Payload Interface,
h. Thermal/Handling/Meteoroid/Debris Protection,
i. ASE

Descriptions of the above elements are given in the following paragraphs and are shown in figure 3.1.2-1.

TABLE 3.1.1-1

OTV STRUCTURAL DESIGN CRITERIA/GUIDELINES

- o MAJOR STRUCTURE DESIGNED TO WITHSTAND LIFTOFF LOADS
  - o N_x = -3.2 G (LIMIT)
  - o N_y = $\pm$1.4 G (LIMIT)
  - o N_z = $\pm$2.5 G (LIMIT)
- o PRIMARY STRUCTURE
  - o ULTIMATE LOAD = 1.5 x LIMIT LOAD

	REUSABLE	
	LH_2	LO_2
o TANKAGE		
o LEAK BEFORE RUPTURE	YES	YES
o MINIMUM FACTOR OF SAFETY TO YIELD	1.5	1.5
o DESIGN SERVICE LIFE (MISSIONS)	45	45
o VENT TO REFILL	YES	PARTIAL
o EQUIVALENT FULL DEPTH CYCLES	55	4
o CONTINGENCY CYCLES	5	5
o SCATTER FACTOR	2	2
o DESIGN CYCLES	120	18
o ROOM TEMP PROOF FACTOR	1.63	1.33

- o METEOROID/DEBRIS SHIELDING
  - o PROVIDE .999 PROBABILITY OF NO TANK WALL IMPACT PER MISSION
  - o MINIMUM METALLIC SHEET IS ASSUMED TO BE .016 in FOR BUMPER

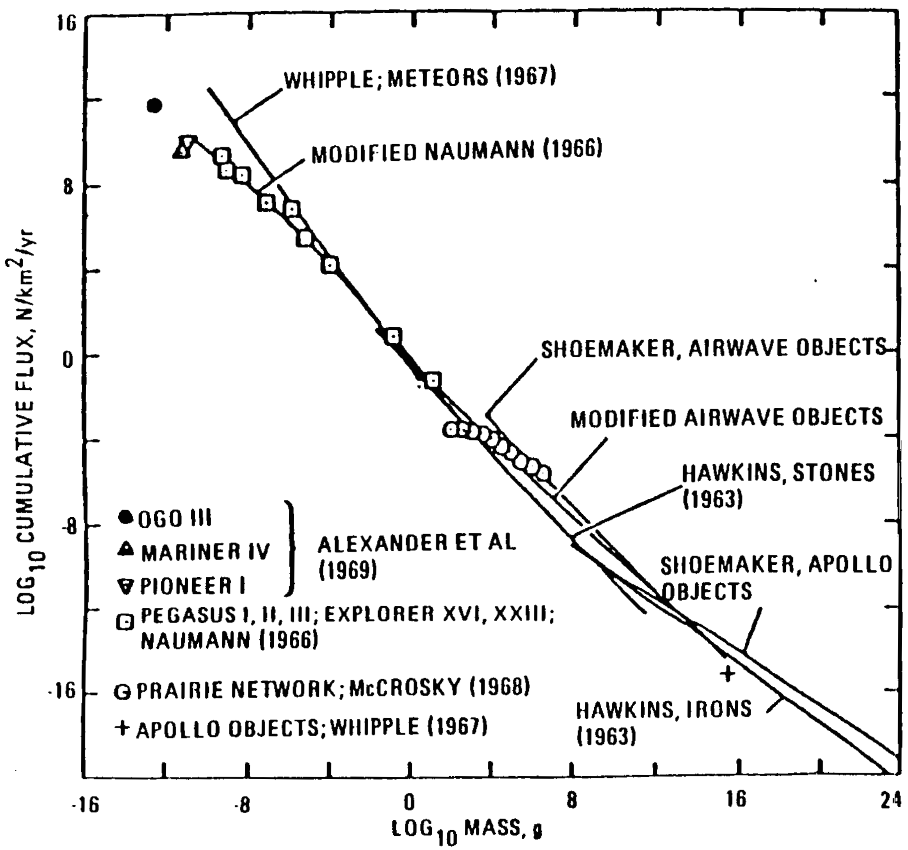
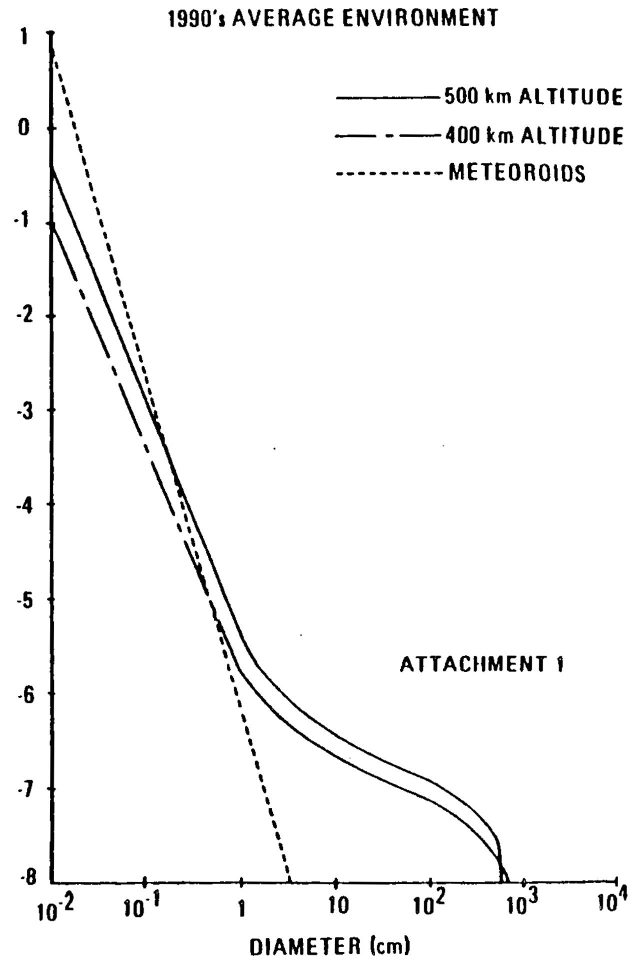


FIGURE 2-13. TERRESTRIAL MASS-INFLUX RATES OF METEOROIDS. N IS THE FLUX OF PARTICLES WITH MASS GREATER THAN m (2-29).

METEOROIDS

NASA TM 82478, "SPACE AND PLANETARY ENVIRONMENT CRITERIA GUIDELINES FOR USE IN SPACE VEHICLE DEVELOPMENT" REV 1982 VOLUME I

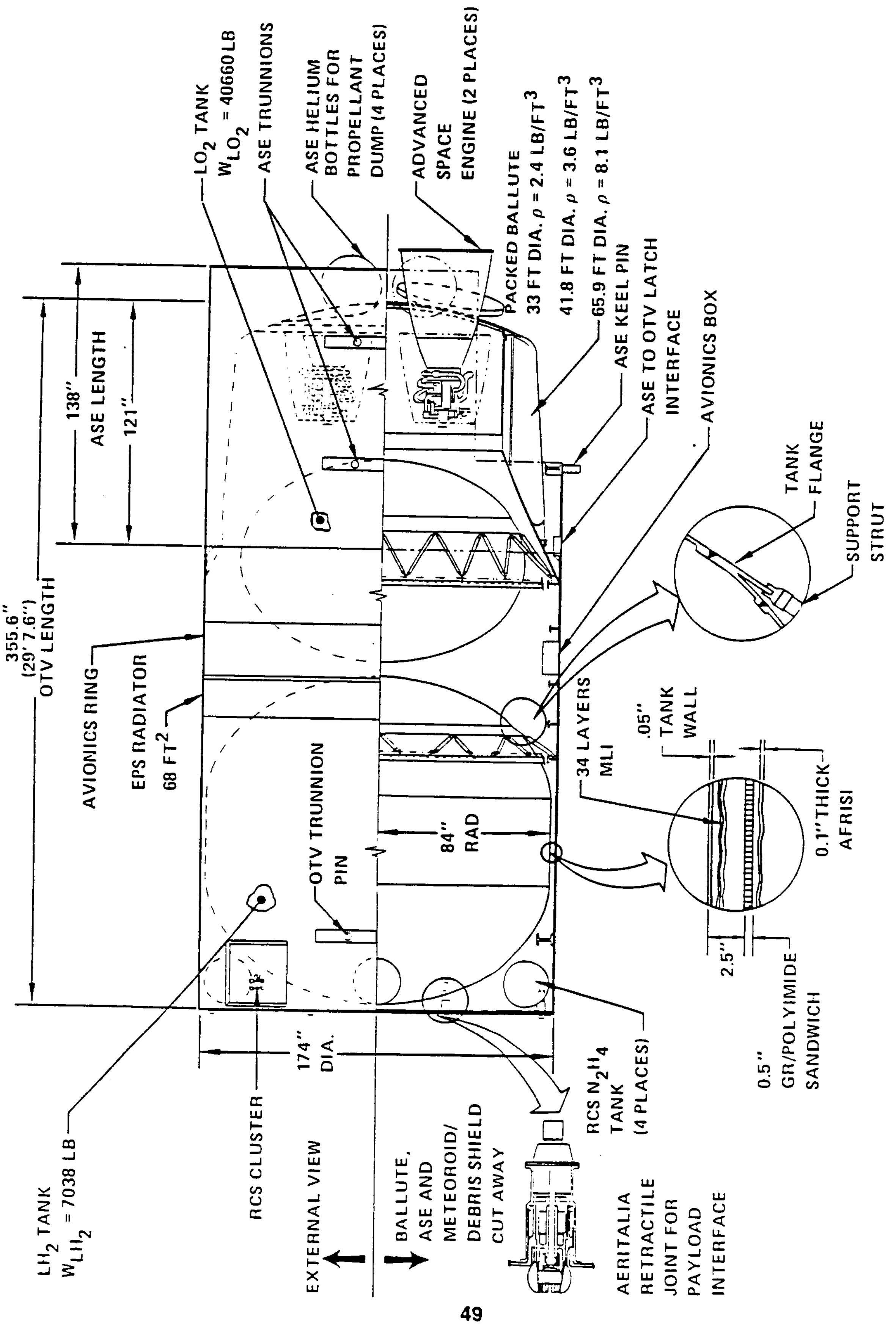


DEBRIS

JSC-20001, "ORBITAL DEBRIS ENVIRONMENT FOR SPACE STATION", D. J. KESSLER, 1984.

Figure 3.1.1-1 Meteoroid/Space Debris Environments

OTV-595



OTV 879

Figure 3.1.2-1 Ground Based Ballute Braked OTV – Main Stage

Aerobrake. For this concept, three sizes of ballutes were designed: 33 ft diameter for unmanned multiple manifest missions, 42 ft diameter for GEO delivery missions when auxiliary tanks are used, and 66 ft diameter manned missions when auxiliary tanks are used and a manned capsule is returned. These are shown in figure 3.1.2-2.

The ballutes are of Nextel fabric gores sewn together and attached to the vehicle at a forward point aft of the avionics/equipment support ring, and at an aft point on the heat shield perimeter. Two hundred and forty meridian straps carry tension loads to the attach points. Structurally, the ballute fabric has a high factor of safety, as shown by the small fabric loading in figure 3.1.2-3. The ballute design shown is a 50 foot diameter ballute, but similar results are found for 33, 42, and 66 foot diameters.

The ballutes are supported during launch, in the packaged condition, around an expendable support cylinder that is external to the heat shield support cylinder.

The ballute support cylinder and heat shield support cylinder are of GR/PI honeycomb sandwich construction, with 3-ply face sheets and 0.5 in, 4 pcf core. The ballute support ring at the heat shield perimeter is GR/PI and has a cross-sectional area of 2.0 in^2.

The central heat shield of the ballute vehicle is a GR/PI honeycomb sandwich structure situated over the main engine compartment, with structural doors to cover the retracted engine nozzles during aeromaneuver. Designed for stiffness and stability at aeromaneuver loading, the sandwich construction has 6-ply face sheets with a 1.0 in thick honeycomb core with a 4 pcf density. The doors are of similar construction. For thermal protection, both heat shield and doors are covered with FRCI-12 rigid thermal tiles.

Thrust Structure. For this concept the thrust structure is a double cruciform, rectangular-beam structure designed to distribute thrust loads evenly to a thrust ring and on to the aft support cylinder. The thrust beam structure is of graphite/epoxy design, with mounting provisions for two engines, including thrust vector controllers. Designed to minimize deflections at the center, assuming stiffness tailoring, the beams were found to have an average cross-sectional area of 4.3 in^2. Additional structure is necessary for assembly and attachment.

Equipment Support Section. The equipment support section is a circular GR/EP structure with aluminum doors for mounting of avionics and electrical power components. Because it is located in the mid-body of the vehicle, this section must also transfer loads. To reduce overall weight, and because the avionics components can be serviced on the ground, the aluminum mounting doors are assumed to be load-carrying.

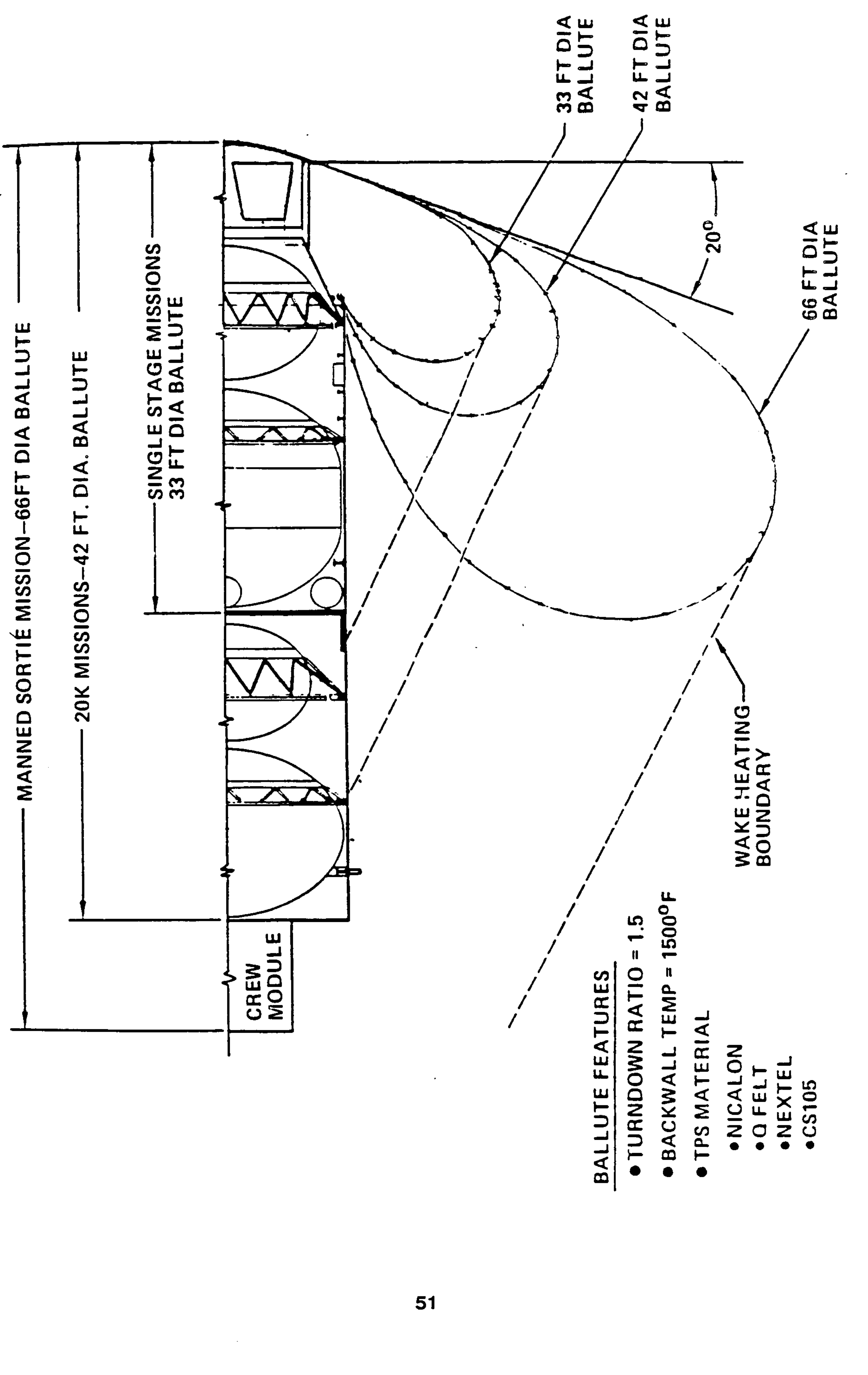
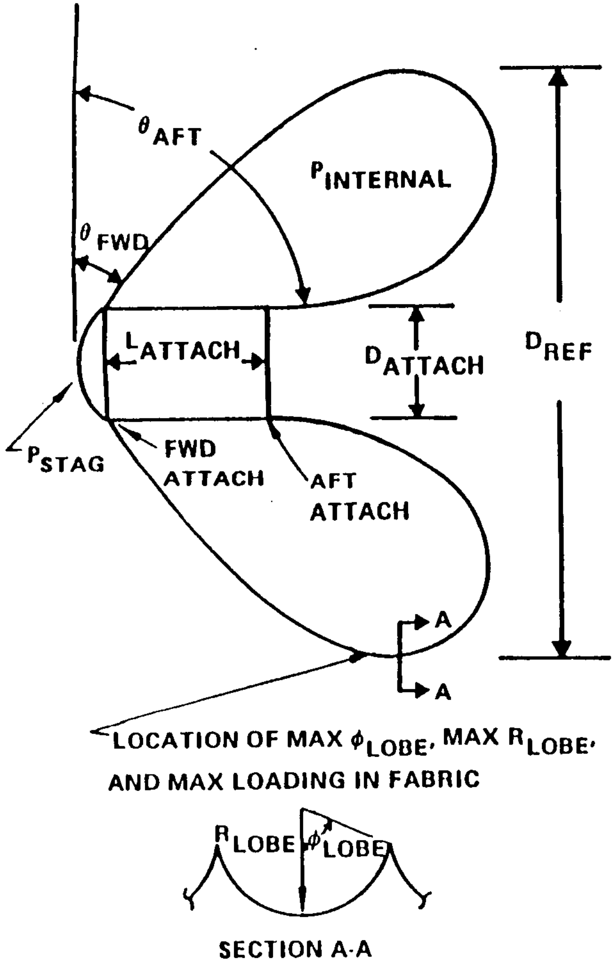


Figure 3.1.2-2 Ballute Aeroassist Provisions — Ground Based OTV



	BALLUTE DESIGN		
	GOODYEAR BASE DATA	DIRECT SCALE-UP	MODIFIED SCALE-UP
D_{REF} (FT)	25.0	50.0	50.0
D_{ATTACH}/D_{REF}	0.28	0.28	0.28
L_{ATTACH}/D_{REF}	0.16	0.16	0.16
$\theta_{FWD\ ATTACH}$ (DEG)	30	30	20
$\theta_{AFT\ ATTACH}$ (DEG)	90	90	90
P_{STAG} (PSI)	0.267*	0.267	0.267
$P_{INTERNAL}/P_{STAG}$	0.78	0.78	0.78
NUMBER OF MERIDIANS	240	240	240
MAX ϕ_{LOBE} (DEG)	88	88	60
MAX R_{LOBE} (IN)	2.0	4.0	4.65
MAX LOADING IN FABRIC (LB/IN)	0.46	0.92	1.07
LOAD IN MERIDIAN (LB)	28	112	112
FABRIC SURFACE AREA (YD^2)	200	800	720
MERIDIAN LENGTH (YD)	9.5	19	19

*CORRESPONDS TO MAX q = 20 PSF

Figure 3.1.2-3 Ballute Structural Fabric Assembly Geometry and Loads

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Propellant Tanks. The propellant tankage is welded 2219-T87 aluminum. The tank shells are sized to permit room temperature proof testing to 1.37 (best-fit fracture mechanics data) times the maximum expected operating pressure (MEOP) of 22.1 psi for the liquid hydrogen tank and 1.32 times the MEOP of 33.3 psi for the liquid oxygen tank.

With respect to the hydrogen tank, the MEOP of 22.1 psi occurs subsequent to the initial OTV main engine ignition, and is based on an ullage maximum vent pressure of 22.0 psi plus a maximum head pressure of 0.1 psig. During operations, the ullage pressure may perturbate several times between 22.0 and 18.0 psia. After return, the tank is purged of its gaseous hydrogen and repressurized with helium before return to the ground. The hydrogen tank average dome thickness is 0.030 in. and cylinder thickness is 0.05 in.

With respect to the oxygen tank the MEOP of 33.3 psi occurs subsequent to shuttle ET burnout during launch, and is based on a propellant vapor pressure of 20.0 psia plus a maximum head pressure of 13.3 psig. During operating, the ullage pressure may perturbate several times between 22.0 and 20.09 psia. After return, the oxygen tank is partially purged and allowed to warm and pressurize before return to the ground. The oxygen tank average dome thickness is 0.031 in.

Due to cryogenic operating conditions, and based on room-temperature proof-testing, the inherent ultimate factor of safety of strength is 2.39 for the hydrogen tank and 1.92 for the oxygen tank.

The propellant tanks are supported within the structural shell by pin ended graphite epoxy struts. They provide structural support and thermal isolation with minimum penetration of the MLI thermal protection system.

Body Structure. The body shell is fabricated from graphite/epoxy facing sheets with a Nomex honeycomb core. The shell is stabilized and reinforced for the various attachments by internal GR/EP rings. This type of structure was chosen for the ground-based vehicle as being the most effective for supporting a payload, as well as the fuel and oxidizer tanks, during launch. Forward of the equipment support section, the shell is made up of 3-ply GR/EP face sheets with a 0.5 in., 4 pcf NOMEX honeycomb core. Aft of this, to the ASE interface, 6 ply face sheets are used with a 0.5 in., 4 pcf core.

The major body rings include a LH_2 tank support ring with 1.0 in^2 cross-sectional area, a LO_2 tank support ring of 1.7 in^2 cross-sectional area, an ASE interface ring of 2.5 in^2 cross-sectional area, and a thrust structure interface ring, of 1.5 in^2 cross-sectional area.

Rings Integral With Tanks. The rings provided to permit the support struts to support the propellant tanks are fabricated integral with and internal to the tanks for structural weight and volume efficiency and to simplify thermal protection.

Typical ring cross-sectional areas vary from 1.5 to 2.0 in^2, for LH_2 and LO_2 tanks, respectively.

Payload Interface. The payload interface is a GR/EP ring fabricated with payload attachment pads and mechanism support, placed at the forward end of the vehicle to support the payload under shuttle launch conditions. It has an average cross-sectional area of 1.5 in^2.

Thermal/Handling/Meteoroid/Debris Protection. Handling and meteoroid/debris protection for this vehicle is provided by the combination of GR/EP sandwich shell and MLI. Due to the relative short duration and nature of the GB OTV missions, additional material does not need to be added to the propellant tanks for meteoroid/debris protection. Based on the analysis provided in Section 3.2.3 of Volume II, Book 3.

For a typical mission, consisting of 18 days in GEO plus 1/2 day in LEO, there is a probability of no tank impact of approximately 0.9998 and a probability of no tank penetration of approximately 0.9999.

ASE Structural Definition. Combined effects of the GB OTV size, operational requirements and the Orbiter payload bay clearance envelopes essentially dictate the ASE design/concept selection. The aft ASE (aft in the Orbiter bay) which support the ballute end of the OTV rotates the vehicle into position for checkout before separation and removal from the Orbiter bay. This dictates the position on the vehicle/ASE and in the Orbiter bay for the aft trunnion. Installation of the ballute prevents the use of vehicle mounted trunnions which necessitates use of a separate aft ASE. The requirement to restow the OTV in the Orbiter bay dictates selection between two options for the vehicle to ASE interface: (1) conventional ordinance separation, such as Super* Zip and separate trunnions and/or latching mechanism for restow, or (2) a linear latch mechanism with relatch capability. Prior investigations of these alternatives indicate that the weight difference between the two options is not significant but the latch mechanism will be more economical for multiple launches.

Return flight and landing of the Orbiter without the OTV necessitates the use of two pairs of trunnions on the aft ASE to maintain the ASE pitch position in the orbiter.

The requirement for launches with the payload attached, and the weight and c.g. location of the payload dictate use of a forward trunnion set to avoid overloading the orbiter by the aft ASE trunnions. Attachment of the payload and the forward OTV diameter dictate the use of forward trunnions and their supporting structure integral with the OTV structure.

Clearances from the LH_2 tank forward head and local variations in the Orbiter attachment point load capabilities dictate the location of the forward trunnion set.

Fabrication and installation tolerances plus Orbiter, ASE and OTV thermal and load induced deflections require that we address the consequences of attaching to the Orbiter through three pairs of trunnions. If all six trunnions were restrained in longitudinal and vertical directions the attachment would be indeterminant to the 7th degree. If not properly addressed this results in unacceptably large loads and weight penalties in the structural system. The longitudinal redundancies are avoided by permitting free motion in the two forward payload retention latch actuators (PRLA) and two of the aft PRLAs. The three vertical redundancies will be addressed by stiffness tailoring of the vertical support structure in the aft ASE to limit the redundant reactions to acceptable values.

3.1.3 Auxiliary Tank Module Structural Definition

The design of the integral auxiliary tank module is conceptually quite similar to that of the GB OTV. A sketch is provided in Fig. 3.1.3-1. The two tank heads and supports for the LH_2 tanks are substantially identical to those for the main stage OTV. The auxiliary LH_2 tank differs from the main stage LH_2 tank only in the elimination of cylindrical section whereas the auxiliary LOX tank is smaller than the main stage LO_2 tank. The structural shell for the module and its support in the Orbiter are quite similar to that for the main stage. The major distinction is a different aft ASE with the keel pin support cantilevered forward of the ASE ring to reduce the yawing moments. The module structural shell is locally recessed to provide clearance for the keel in support. The clearance available for the aft ASE, and not available for the main stage aft ASE permits a much stiffer and lighter ASE structure. The forward trunnion support is substantially identical to that for the main stage.

3.1.4 Weight Summary

Detail weights for the GB OTV main stage and auxiliary tankset structures are presented in Table 3.1.4-1. These include weights for a 33 ft. diameter ballute.

Weights for the jettisonable portions of the three different size ballutes (33 ft, 42 ft, 66 ft) are given in Table 3.1.4-2.

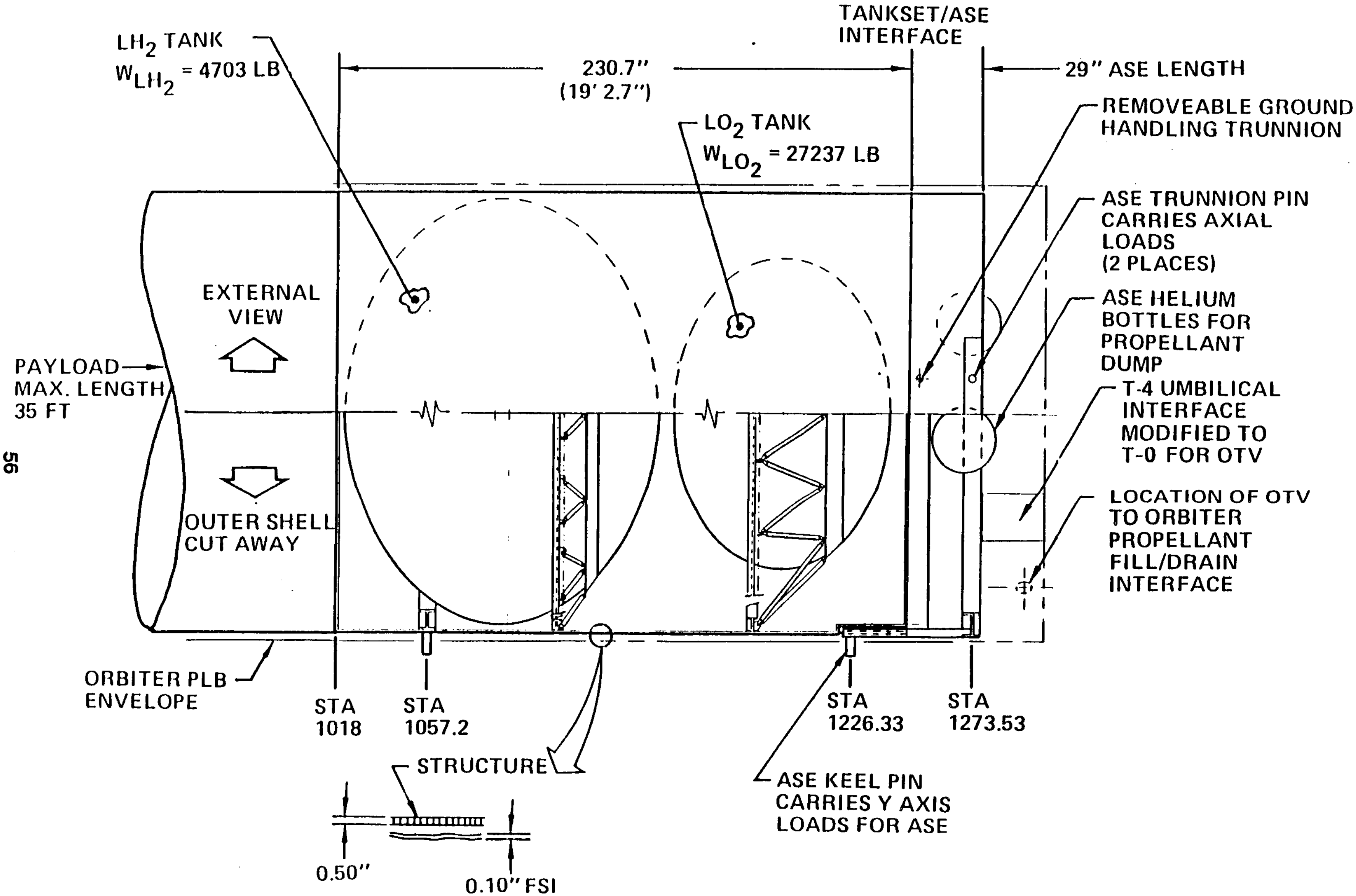


Figure 3.1.3-1 Auxiliary Tankset Ground Based Ballute Braked OTV

Table 3.1.4-1. Structures Detail Weight Statement

	Main Stage	Auxiliary Tanks
Basic Structure	(1629)	(973)
Body Shell	546	421
Support Rings	319	160
Thrust Structure	186	
Equipment Mounting Structure	240	
Payload Interface	51	51
Umbilical Interface	25	40
Orbiter Attach. Ftgs.	90	90
Grapple Ftgs.	22	22
Berthing Probes	20	20
Payload Mechanisms-Latch/Release	96	144
Misc Assy/Support Hardware	34	25
Tanks	(993)	(757)
Fuel Tank	556	261
Fuel Tank Support	32	32
Oxidizer Tank	331	392
Oxidizer Tank Support	74	72
Aeroassist Device (33 ft. dia)	(983)	(N/A)
Ballute-Jettisonable	316	
Support Structure-Jettisonable	229	
Support Structure-Non-Jettisonable	365	
Inflation Provisions	73	
TOTAL	3605 lbm	1730 lbm

TABLE 3.1.4-2

GB BALLUTE WEIGHTS (lbm)

Item	Multiple Payload Delivery	20k Delivery	Manned 7.5k Roundtrip
Diameter	33 ft.	42 ft.	66 ft.
Ballute Structure-Jettisonable	(545)	(725)	(1404)
Fabric Structure	316	479	1100
Support Structure	229	246	304
Thermal Protection-Jettisonable	(264)	(373)	(796)
Flexible	248	357	780
Rigid	16	16	16
Weight Growth	(121)	(165)	(330)
TOTAL JETTISONABLE WEIGHT	930	1263	2530

3.2 PROPULSION

OTV propulsion systems include the MPS (main propulsion system) and the RCS (reaction control system). The MPS is described and discussed in section 3.2.1 and the RCS is described and discussed in section 3.2.2.

3.2.1 Main Propulsion System

3.2.1.1 Top Level Requirements

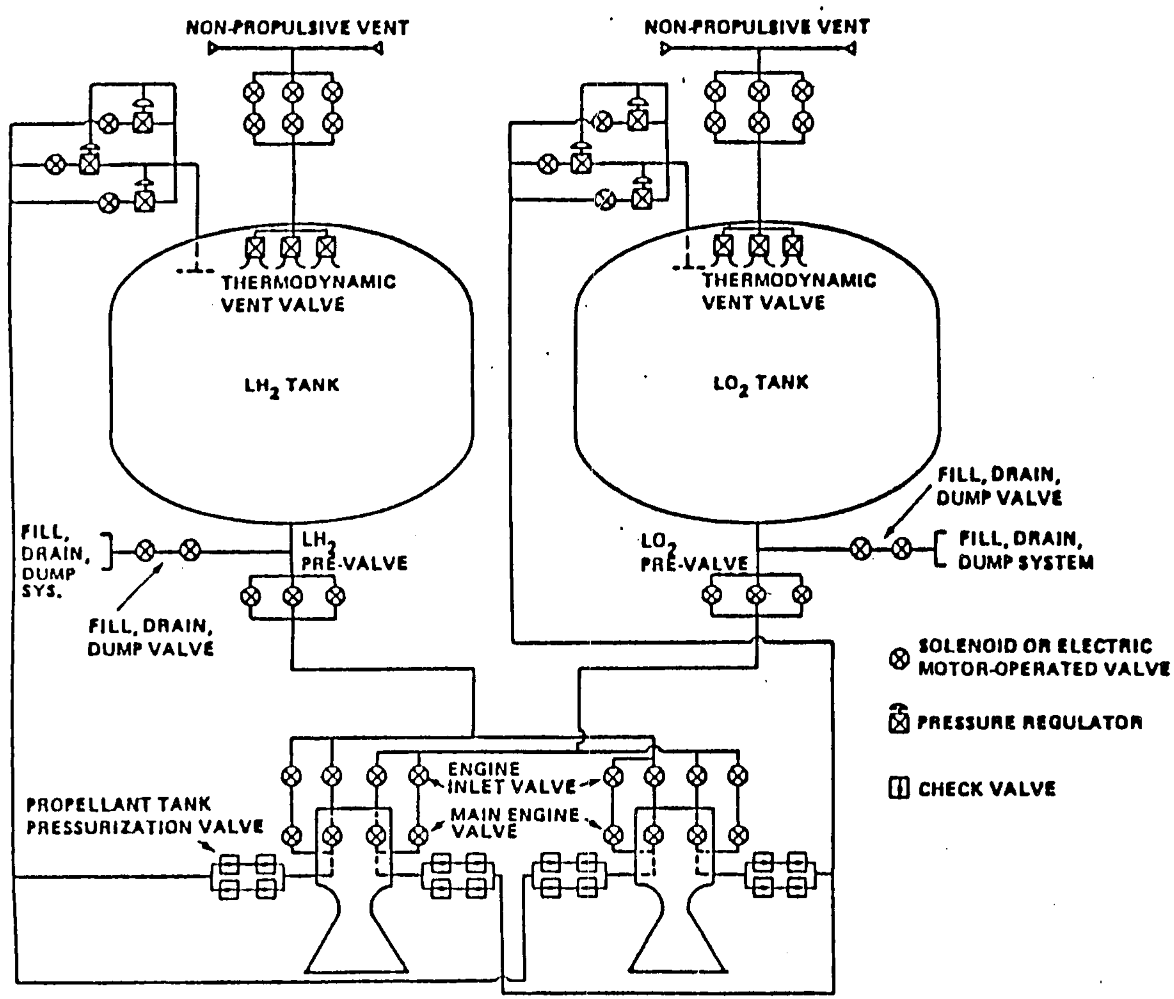
The OTV is planned to reduce the costs of accomplishing a variety of upper stage missions now performed by storable propellant and solid rocket motor stages. Payloads exceeding the capabilities of existing stages are also planned as well as manned missions. Top level requirements derived from these objectives are:

a. Provide thrust for delta-velocity maneuvers required for geosynchronous and other high energy maneuvers.
b. Be re-useable for at least 10 missions to minimize recurring costs.
c. Satisfy man rating requirements.
d. Be capable of operating in either a ground based or space based mode.
e. Be compatible with shuttle launch capability.

3.2.1.2 Hardware Definition

This study resulted in selection of advanced engines using cryogenic oxygen and hydrogen propellants. The schematic shown by figure 3.2.1-1 identifies the propulsion system functional components required to support all flight operations of the engines. Two advanced cryogenic oxygen/hydrogen engines and redundant systems valving are included to support man rating.

Advanced engines similar to those defined by Aerojet were used for the performance baseline (does not imply engine recommendation). The engine's valves and all system valves are electrically powered eliminating any requirement for a pneumatic helium system. An oxygen turbine is used to power the oxidizer pump and a hydrogen turbine is used to power the fuel pump, therefore helium is not required to purge pump seals. Rocketdyne and Pratt & Whitney engines use hydrogen turbines to power the oxygen pumps and will require a small helium supply unless a vacuum purging system for the seals is developed. The engines are capable of starting with zero NPSH and with either liquid or vapor at the interface. This capability will be used to settle propellants; therefore, a low "g" propellant acquisition device is not required in the tanks for flight operations. The engines include provisions for supplying autogenous pressurization for the tanks.



- DUAL FAILURE TOLERANT DUAL ENGINES
- HIGH RELIABILITY (0.99573)
- AT LEAST FAIL SAFE IN ALL COMPONENTS

Figure 3.2.1-1 Baseline Main Propulsion Subsystem

OTV-1716

The propellant tanks are of aluminum construction with MLI insulation. The tanks structure is described in paragraph 3.1.2 and a weight statement is included. The propellant utilization and gaging system (PUGS) is a capacitance probe which provides propellant quantity data during thrusting periods when the propellants are settled. Baffles are included to reduce liquid settling times and prevent liquid from overshooting or bypassing the outlet during settling. Thermodynamic vent valves of the type currently being developed for the Centaur will be used for venting required during low "g" periods without settled liquids. A liquid acquisition device has not been included in the tanks.

3.2.1.3 Function Description

The main propulsion system (MPS) will be used for all delta-velocity maneuvers greater than 20 ft/s. Typical delta-velocity requirements for a geosynchronous payload delivery mission are:

a. Two-burn transfer from LEO to GEO.
   1. First burn is 3649 ft/s
   2. Second burn is 4459 ft/s
b. Circularization at GEO is 5798 ft/s.
c. Transfer to LEO burn is 6245 ft/s.
d. Orbit correction for aeromaneuver is 50 ft/s.
e. Orbit correction after aeromaneuver is 251 ft/s.
f. Circularization for recovery burn is 420 ft/s.

Each burn of the main propulsion system will be accomplished with the following sequence of events.

a. Check system status and verify readiness.
b. Open pre-valves to bleed through engines, cooldown and settle propellants.
c. Bring engines to pumped idle mode.
d. Pressurize tanks for full thrust operation.
e. Accelerate engines to full thrust and complete delta-velocity maneuver.
f. Shut down engines.
g. Close pre-valves and bleed lines through engines.
h. Close engine valves and verify status.

During coasting periods the thermodynamic vent valves will vent the propellant tanks as required to maintain tank pressures in acceptable limits. The vent valves will function individually at pressure settings approximately one psi apart. The sequential pressure settings will cause only one valve to function at any time to limit the heat input to the tanks caused by the mixer. Three valves are provided for dual failure tolerance of the critical vent function.

3.2.2 Reaction Control System

3.2.2.1 Top Level Requirements

The reaction control system is used to control the vehicle orientation during coasting periods and perform maneuvers which do not warrant use of the main propulsions system. Top level requirements to support the OTV missions and objectives are:

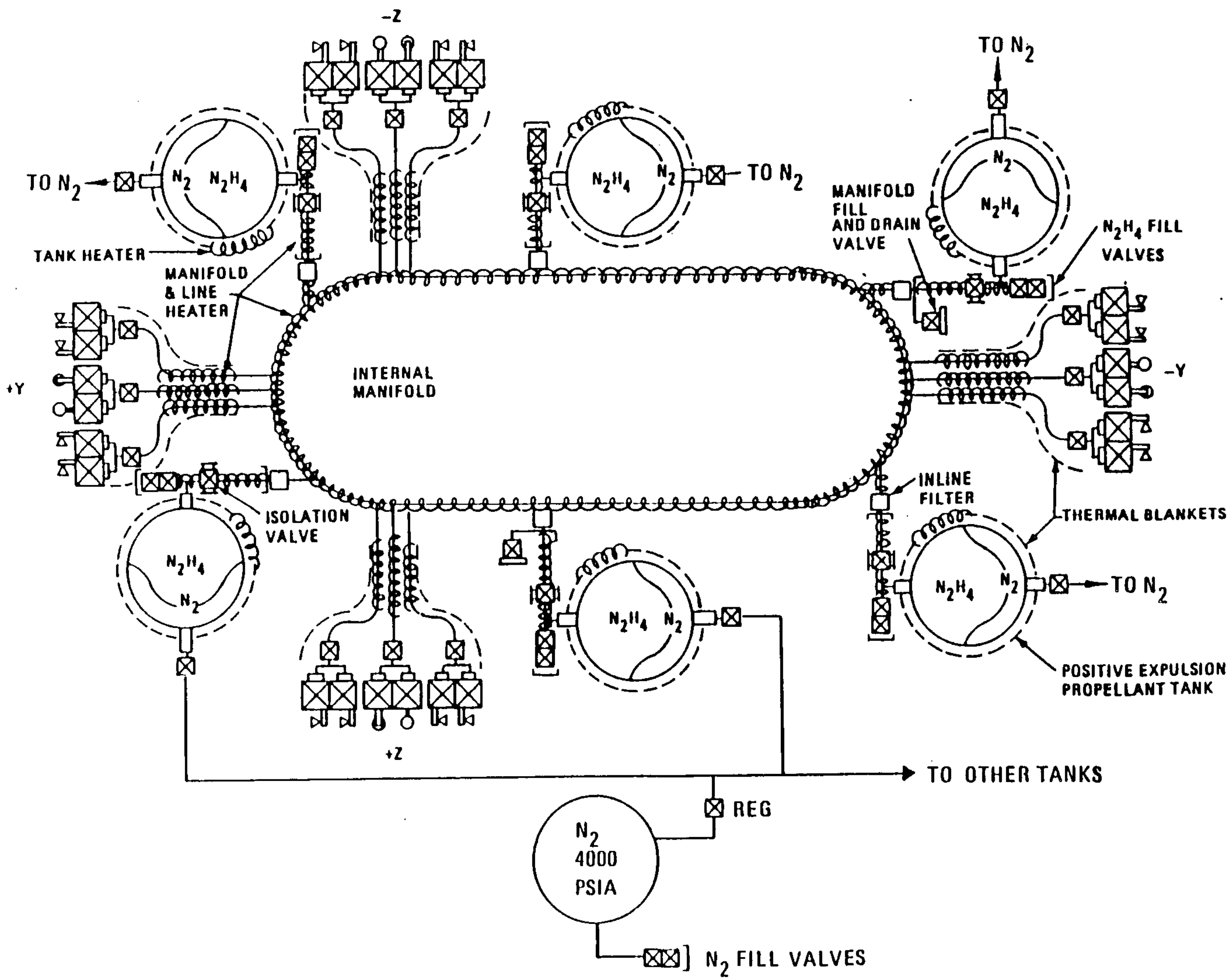
a. Provide thrust for delta-velocity maneuvers of less than 20 ft/s.
b. Be reuseable for at least 20 missions to minimize recurring costs.
c. Satisfy man rating requirements.
d. Control orientation of the vehicle and provide initial pointing for main propulsion system start.
e. Be capable of operating in either a ground based or space based mode.
f. Be compatible with shuttle launch.
g. Provide six degree of freedom control for docking maneuvers.

3.2.2.2 Hardware Definition

The reaction control system selected during this study is a monopropellant hydrazine system with pressurized propellant tanks. The functional arrangement of the thrusters, tanks and other components is shown in figure 3.2.2-1. Redundancy in six degrees of freedom is obtained from 24 thrusters (6 thrusters per module) during manned missions for dual failure tolerance. Reduced redundancy is acceptable for missions which are not manned and only 16 thrusters (4 thrusters per module) will be installed for these missions.

Thrusters are arranged in four clusters with six thrusters each. Two thrusters point aft, and two thrusters point to each side. Steady state thrust of each thruster is approximately 30 lbf at the maximum operating pressure of 380 psia. The minimum impulse bit is 0.30 lbf-sec. Thrusters are provided with thermostatically controlled electrical heaters to prevent propellant freezing.

The propellant tanks contain bladders to separate the hydrazine from the nitrogen pressurizing gas and provide positive liquid expulsion. Each tank is connected to a manifold to distribute the propellant to the thruster clusters. Thermostatically controlled electrical heaters maintain the tanks, manifold and connecting lines above the freezing temperature of the hydrazine propellant. Six tanks are used because placement



- 6 DEGREE OF FREEDOM CONTROL
- COMPLETE DUAL REDUNDANCY
- NITROGEN PRESSURIZATION
- 24 THRUSTERS
- 25 LBF THRUST
- 1500 LBM HYDRAZINE PROPELLANT

Figure 3.2.2-1 Baseline Reaction Control Subsystem

OTV-851

within the available envelope was found to be difficult with a fewer number of larger tanks.

3.2.2.3 RCS Function Description

The reaction control system is used during coasting periods of the mission. The system is active and responds to impulse bit commands from the guidance system to maintain vehicle attitude. Other mission functions performed by the RCS operating in a steady state mode include:

a. Provide thrust for orbit corrections requiring less than 20 ft/s delta-velocity.
b. Separate the OTV from payloads.
c. Provide vehicle translation and attitude control for docking and during shuttle or Space Station rendezvous.

The RCS will be maintained in an inert condition when docked to the Space Station or in the Shuttle payload bay. During the inert condition the thrusters modules do not contain propellant and are isolated from the manifold and tanks by valves provided for that purpose. After separation from the Shuttle or Space Station the isolation valves are opened to charge the system with propellant. The nitrogen gas pressure regulator maintains the tanks pressure during the mission to provide thruster interface pressure requirements. The system is returned to the inert condition prior to docking with the Shuttle or Space Station by closing the isolation valves and opening the thruster valves to dump propellants from the thruster modules.

3.2.3 Detail Weights

Detail weights for the GB OTV propulsion systems, including main propulsion and reaction control, are presented in Table 3.2.3-1. Shown are weights for both unmanned cost optimum and manned configurations of the main stage, as well as plumbing weights for the auxiliary tankset.

Table 3.2.3-1. Propulsion Systems Detail Weight Statement

	Main Stage		Auxiliary Tank
	Unmanned Cost Optimum	Man-Rated	
Main Propulsion System (MPS)	(1003)	(1003)	(410)
Engines (2)	396	396	N/A
Thrust Vector Control	45	45	N/A
Tank Pressurization	101	101	50
Feed/Fill/Dump-Fuel	148	148	112
Feed/Fill/Dump-Oxidizer	143	143	100
Vent/Relief-Fuel	104	104	93
Vent/Relief-Oxidizer	66	66	55
Reaction Control System (RCS)	(248)	(345)	(N/A)
Thrusters (16)	43	65	
Tankage	122	197	
Tank Pressurization	41	41	
Manifold, Plumbing	42	42	
TOTAL	1251 lbm	1348 lbm	410 lbm

3.3 THERMAL PROTECTION AND CONTROL

3.3.1 Top Level Requirements

Thermal protection systems (TPS) must:

a. Provide an aerodynamic surface capable of operating while subjected to the aerothermal environments associated with the aeropass maneuver.

b. Protect the primary structure from effects of the aerothermal environment.

The worst case thermal design environments are shown in table 3.3-1 for the three baseline ballute concepts, which shows design heating rates up to 31.5 BTU/ft^2-sec on its flexible surface insulation (FSI), and up to 45.8 BTU/ft^2-sec on the rigid surface insulation (FSI) surface.

Other requirements imposed on the TPS include:

a. reusability or easy replacement,

b. capability of being assembled or deployed in orbit, and

c. light weight.

The thermal control system must provide a means of dissipating heat generated by the avionics unit, and also of protecting the avionics components from the aerothermal environment during the aeropass maneuver.

3.3.2 TPS Hardware Definition

TPS concepts for the GB SCB vehicle include thermal insulation on the ballute, on the heat shield, over sensitive avionics installations, and around the vehicle itself. The ballute is made of Nextel fabric (ceramic fibers) and is coated on the inside with CS105—a glass-frit-impregnated high temperature resin—as a gas barrier.

Candidate TPS concepts selected for the baseline OTV configurations are illustrated in figure 3.3-1. TPS concepts and insulation thicknesses for the 33 and 66 foot ballute configurations are defined in figures 3.3-2 and 3.3-3, respectively. The baseline TPS consists of:

a. a 0.2 inch quilted FSI blanket on the windward surface of the 33 foot ballute, and 0.1 inch on the 66 foot ballute,

b. a 0.375 inch strip of quilted FSI blanket on the base of the ballute adjacent to the avionics ring,

c. a 0.1 inch quilted FSI blanket over the exposed OTV structure behind the ballute,

d. a 0.42 inch quilted FSI blanket over the OTV structure exposed to the inside of the ballute, and

Table 3.3-1 Baseline Aerothermal Environment and TPS Weight Summary

GROUND BASED 1500°F BALLUTE

MISSION	DIAMETER (FT)	REENTRY WEIGHT (LBS)	W/C_DA (PSF)	q_o (BTU/ FT^2-SEC)	Q_o (BTU/FT^2)	q_{MAX} (BTU FT^2-SEC)		TPS WEIGHT (LBS)	
						FSI	RSI	FSI	RSI
UNMANNED MULTI MANIFEST	33	10,926	12.05	128	16,700	31.5	45.8	626	125
UNMANNED 20K PAYLOAD	40	13,900	10.43	103	15,550	29.3	42.6	722	125
MANNED 75K RETURN	66	23,934	6.60	119	13,050	25.4	36.8	1244	125

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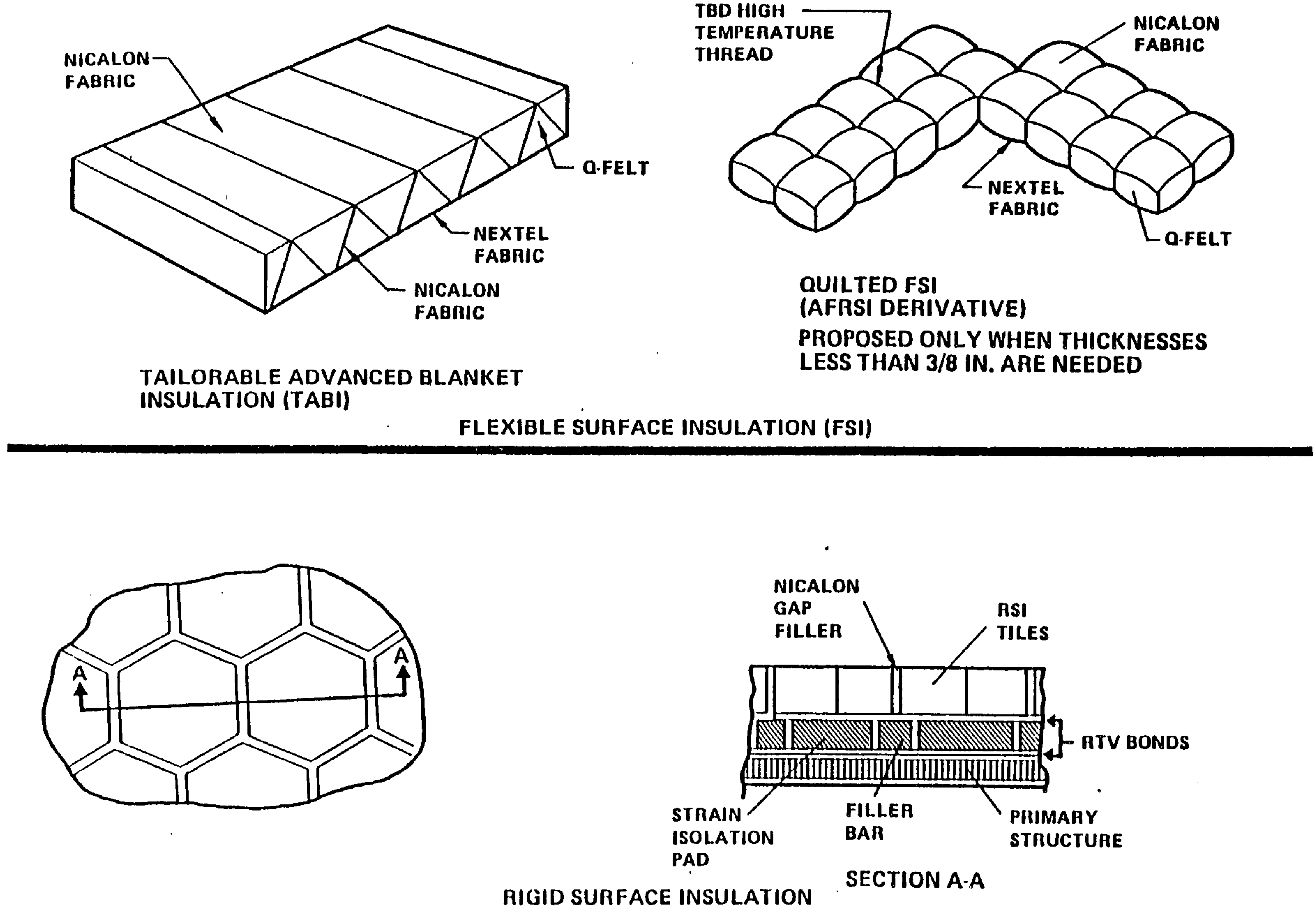


Figure 3.3-1 TPS Concepts

OTV-642

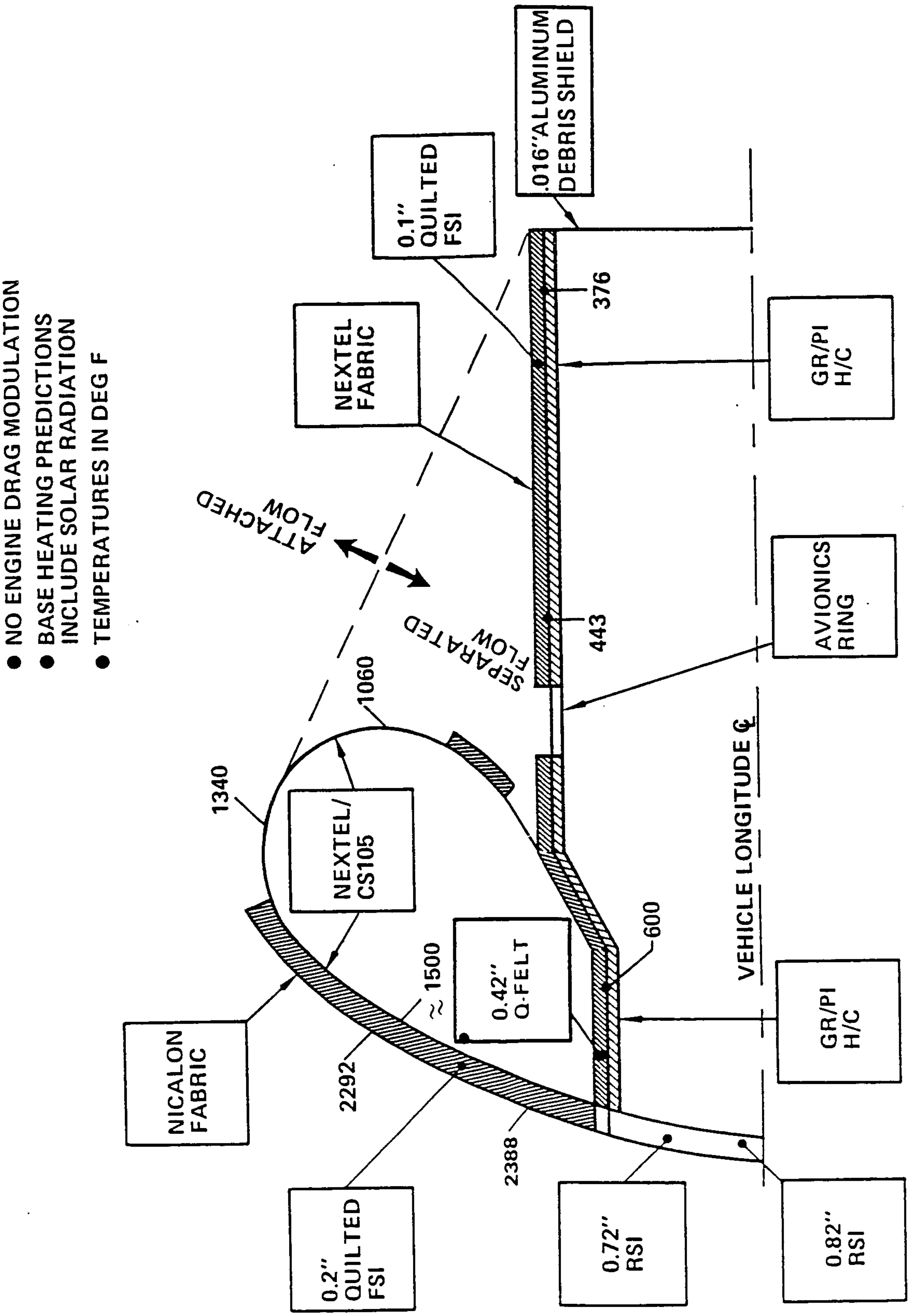


Figure 3.3-2 Maximum Temperatures on a 33 Ft. Ground Based Ballute

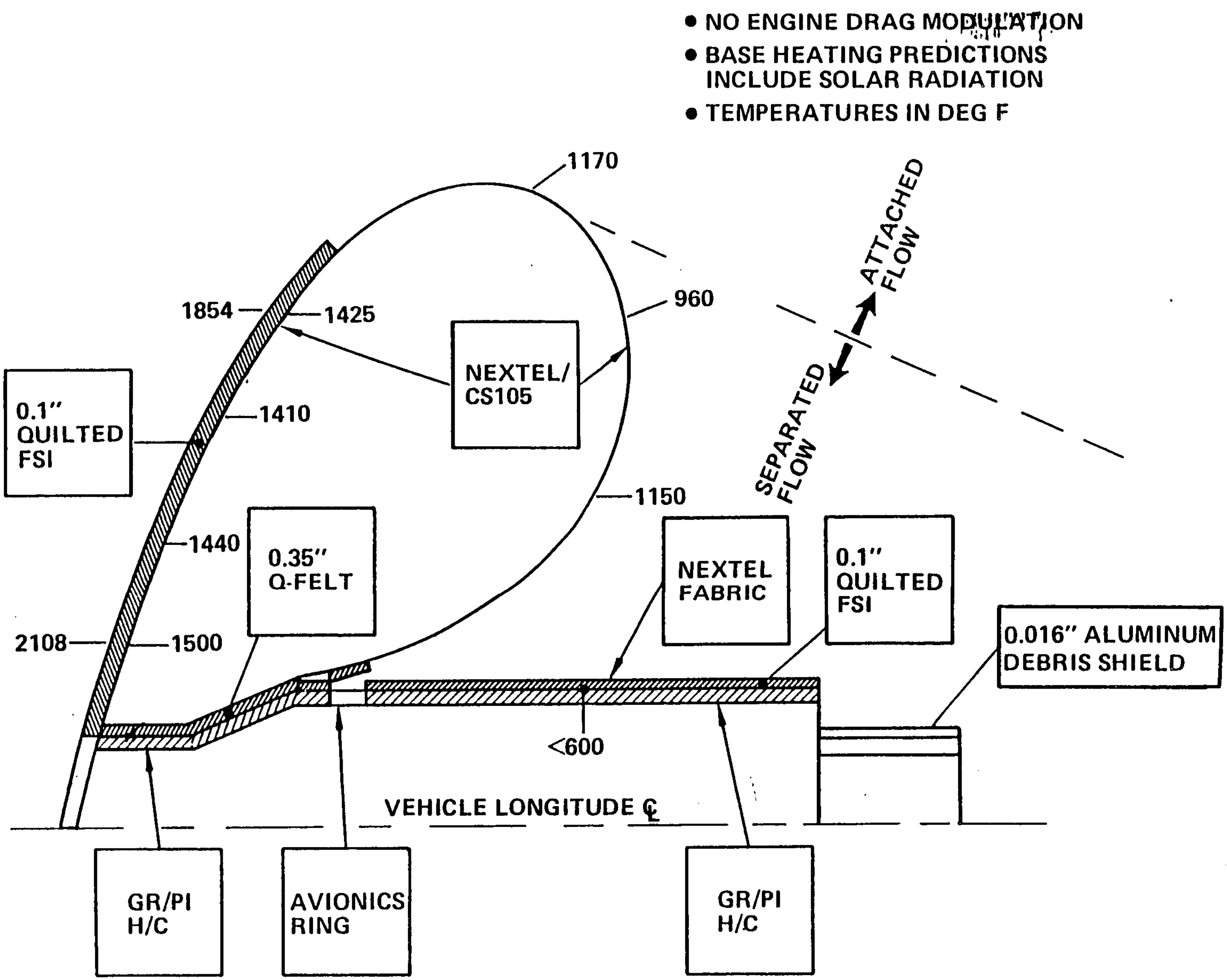


Figure 3.3-3 Maximum Temperatures on a 66 Ft. Ground Based Ballute

OTV-897

e. aluminum enhanced thermal barrier (AETP) tiles bonded to a Nomex strain isolation pad over the aeroshell surface.

Insulation thicknesses were selected to insure that the Nextel/CS105 gas barrier does not exceed 1500°F, and that the graphite/polyimide OTV structure does not exceed 600°F. The FSI and RSI selections given above are tentative. TABI is superior to the quilted FSI in several ways, but currently cannot be fabricated in thicknesses less than about 3/8 inches. TABI would be substituted for the quilted blankets if thicknesses down to 0.1 inch could be obtained. Also, AETP appears to have the potential for meeting RSI requirements for OTV, but is still in the development stage (reference 3-1). Alternate materials, such as high temperature performance (HTP, reference 3-2) would be substituted if AETP does not perform as expected or if alternate materials appear better suited for OTV.

3.3.3 Detail Weights

Detail weights for the GB OTV thermal protection and control system are presented in Table 3.3-2. Shown are weights for both the main stage and the auxiliary tankset.

Table 3.3-3 gives the jettisonable portion of the thermal protection for the three sizes of ballutes used with this vehicle.

Table 3.3-2. Thermal Protection and Control Detail Weight Statement

	Main Stage	Auxiliary Tanks
Aeromaneuver insulation (33 ft. dia)	(650)	(100)
Ballute-Jettisonable	264	
Rigid-Non-Jettisonable	93	
Vehicle Body-Non-Jettisonable	293	100
Propulsion System Thermal Control	(257)	(175)
Tank Insulation	147	110
Other Insulation	21	15
Purge Provisions	90	50
Electrical Power Thermal Control	(87)	(N/A)
TOTAL	994 lbm	275 lbm

TABLE 3.3-3

GB BALLUTE WEIGHTS (lb)

Item	Multiple Payload Delivery	20k Delivery	Manned 7.5k Roundtrip
Diameter	33 ft.	42 ft.	66 ft.
Ballute Structure-Jettisonable	(545)	(725)	(1404)
Fabric Structure	316	479	1100
Support Structure	229	246	304
Thermal Protection-Jettisonable	(264)	(373)	(796)
Flexible	248	357	780
Rigid	16	16	16
Weight Growth	(121)	(165)	(330)
TOTAL JETTISONABLE WEIGHT	930	1263	2530

3.4 GUIDANCE AND NAVIGATION

The guidance and navigation subsystem provides the means to determine the flight path of the vehicle throughout the mission.

3.4.1 Top Level Requirements

The top level requirements for the guidance and navigation subsystem are:

a. Provide vehicle attitude determination.

b. Provide vehicle position data.

c. Provide vehicle velocity status.

3.4.2 Hardware Definition

The guidance and navigation (G&N) subsystem consists of a ring laser gyro inertial reference unit (IRU), a star tracker, and a global positioning system (GPS) receiver-processor and antenna installation. The IRU and GPS are internally redundant. An additional star tracker is added for manned missions. The equipment list for G&N is shown in table 3.4-1. The functional relationship to other avionic elements is presented in figure 3.5.2-1 of Section 3.5.

3.4.3 Functional Description

The ring laser gyro IMU provides angular rate and linear acceleration data. Attitude is initialized and updated by the star tracker. Position and velocity measurements are provided through use of the NAVSTAR GPS via the on-board GPS receiver/processor and antenna.

TABLE 3.4-1

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM EQUIPMENT LIST

	Dimensions (in)			Unit	Unit	Quantity
Subsystem Element	L	W	H	Mass (Lb)	Power (Watts)	Vehicle
Inertial Reference Unit	16.2	11.2	5.9	35	70	2
Star Tracker	10.0	6.5	6.5	20	18	1
GPS Receiver/Processor	11.5	8.0	8.2	23	57	1
GPS Antenna	4.1	4.6	1.5	0.5	Nil	2
Hybrid Coupler		TBD		1.0	Nil	1
Totals				115	215	

Note: For manned mission add one star tracker.

3.5 COMMUNICATIONS AND DATA HANDLING

The vehicle communications and data handling subsystem provides all computation, monitoring and control of the vehicle and its subsystems and provides the communications between the vehicle and all Earth and orbital support elements. The following paragraphs provide the top level subsystem requirements, definition of the hardware, and a functional description for the communications and data handling subsystem.

3.5.1 Top Level Requirements

The top level requirements for the communications and data handling subsystem are:

Communications

a. Provide telemetry, tracking, and communications between the vehicle and other support elements. These elements include the Orbiter (hardline through the ASE and RF), the Space Station (hardline and RF), ground (hardline through Orbiter umbilicals and RF), and TDRS (RF).
b. Provide ranging signal turnaround for both TDRS and GSTDN.
c. Provide components to achieve the cost optimum unmanned configuration with capability to incorporate additional equipment for a dual failure tolerant man-rated configuration.
d. Provide a telemetry, tracking and command transmission capability compatible with STDN and TDRS when in flight outside the Orbiter.

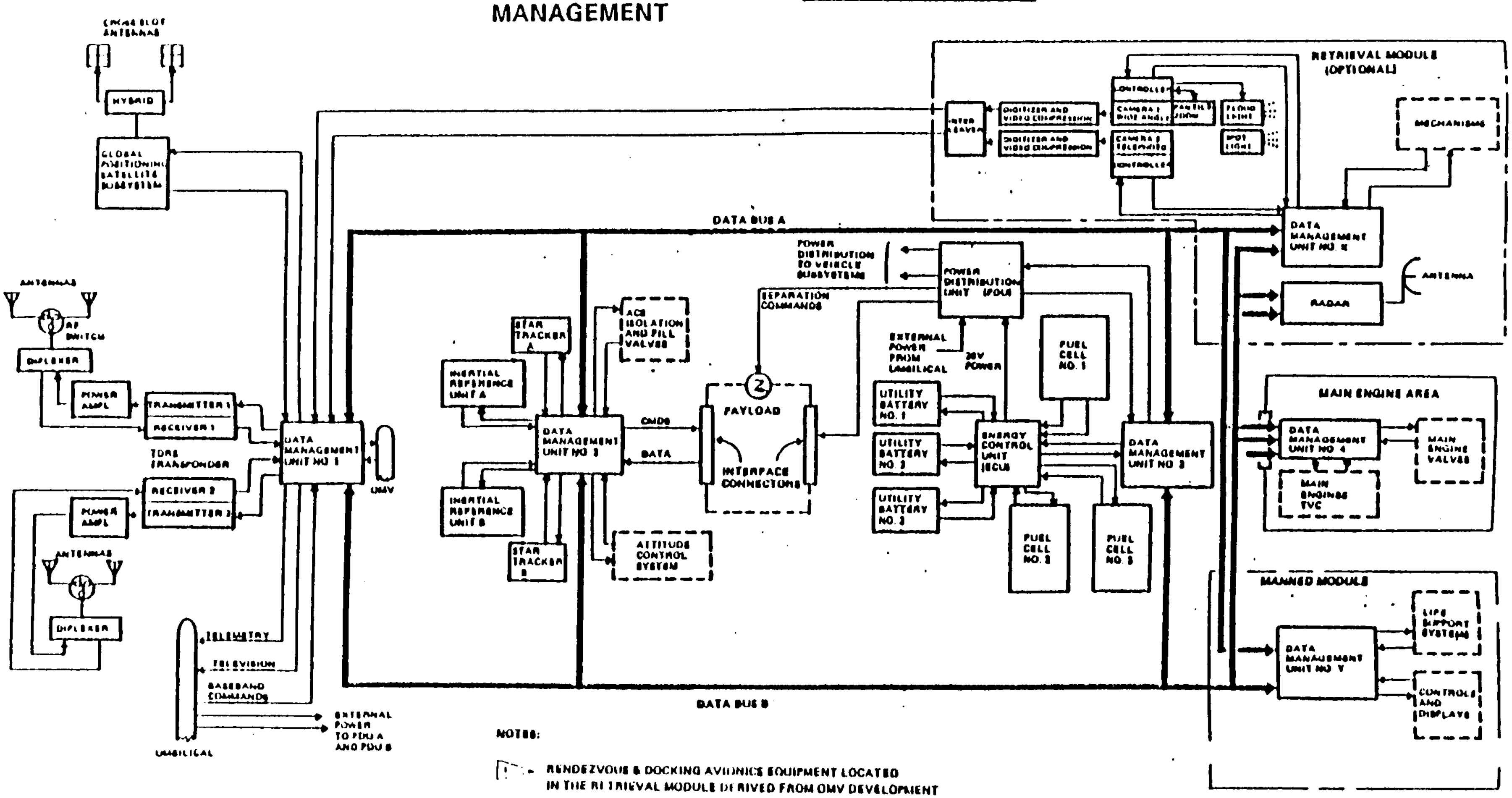
Data Handling

a. Provide measurement of the status of vehicle subsystems. Acquire the data, condition it as required, format it, and provide it to telemetry and to the software for computation as required.
b. Perform computational tasks for all vehicle subsystems and vehicle GN&C.
c. Provide built-in-test capability to isolate failures to LRU.
d. Perform vehicle automatic checkout.
e. Provide redundancy management.
f. Provide components necessary for cost optimum unmanned vehicle and man-rated vehicle.

3.5.2 Hardware Definition

The overall avionics schematic for the manned mission is shown in Figure 3.5.2-1. As discussed in Section 10.0 (Reliability/Safety) of Volume II, Book 3 (Vehicle Level Analysis), an analysis was performed to determine the most cost effective approach to

- HIGH SPEED SERIAL DATA BUS
- FULLY AUTONOMOUS REDUNDANCY MANAGEMENT
- MISSION CRITICAL COMPONENTS ARE FAIL OP/FAIL SAFE
- DUAL REDUNDANT SYSTEM



NOTES:

- RENDEZVOUS & DOCKING AVIONICS EQUIPMENT LOCATED IN THE RETRIEVAL MODULE DERIVED FROM OMV DEVELOPMENT
- THE MANNED MODULE WILL HAVE AUTONOMOUS AVIONICS WITH INTERFACE TO OTV AVIONICS VIA THE DMS DATA BUSES

Figure 3.5.2-1 Recommended Avionics Subsystem

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the avionics configuration for the OTV program considering both manned and unmanned missions. The results of the analysis showed that it was more cost effective to configure the avionics initially with some redundant elements, but not total redundancy. Included in this initial avionics configuration was the equipment required to accommodate extra avionics for the manned mission. Table 3.5.2-1 provides a listing of the communications and data handling components for the cost optimum configuration.

For the manned mission configuration, the equipment complement of the cost optimum unmanned subsystem is supplemented to achieve the required redundancy level. The supplemental equipment includes an additional communications link and the installation of additional printed circuit cards and power supplies within the data management units to achieve the required internal redundancy for the manned mission. The equipment list of the communication and data handling subsystem for the manned mission is shown in Table 3.5.2-2. This list when compared to the cost optimum list of Table 3.5.2-1 demonstrates the additional equipment quantities, increased weights, and power consumptions for the manned missions.

The transponders are STDN/TDRS compatible and operate in the S-band. The RF amplifiers are solid state and provide 20 watts minimum at the S-band operating frequency. The omni antennas are conical log spiral and are installed on diametrically opposite sides of the vehicle.

The data management unit (DMU) design is derived from the Boeing IR&D effort to develop an Integrated Fault Tolerant Avionics System (IFTAS). The IFTAS design features the MIL-STD 1750A architecture and internal redundancy. The DMU's are connected to both of the fiber optic data busses. Each DMU contains internal processors and memory, input/output circuitry, and built-in-test capability to isolate failures to the printed circuit card level. The trades and analyses leading to the selection of the IFTAS design as the basis for the vehicle data handling subsystem are discussed in Section 7.0 of Volume II, Book 3, "Vehicle Analysis."

3.5.3 Functional Description

The communications and data handling subsystem provides for all communications (hardline and RF), data acquisition, data processing, signal conditioning, command outputs, mission sequencing, subsystem status monitoring, and redundancy management functions required to accomplish OTV mission objectives. The following paragraphs contain functional descriptions for vehicle communications, data handling, and instrumentation.

TABLE 3.5.2-1

COST OPTIMUM COMMUNICATIONS AND DATA HANDLING EQUIPMENT LIST

UNMANNED MISSIONS

Subsystem Element	Dimension(in.) L	W	H	Unit Mass(Lb.)	Unit Power(watts)	Quantity Per Vehicle
Communications						
Transponder	14.3	6.0	4.8	13.9	36.0	1
RF Power Amplifier	11.0	13.0	5.0	13.0	110.0	1
Diplexer	7.3	2.4	3.4	2.0	Nil	1
RF Switch	2.7	4.0	1.2	0.6	Nil	1
Omni Antenna	14.6	5.6	(Cone)	1.5	Nil	2
Cabling	N/A	N/A	N/A	10	Nil	N/A
Data Handling						
Data Management Unit (DMU) #1	16.9	8.0	6.0	28.0	99.8	1
DMU #2	28.8	8.0	6.0	50.0	180.6	1
DMU #3	25.0	8.0	6.0	45.0	169.8	1
DMU #4	18.3	8.0	6.0	33.0	134.6	1
Data Bus	N/A	N/A	N/A	30.0	Nil	N/A
Instrumentation						
LO_2 Propellant Sensors	N/A	N/A	N/A	15.0	Nil	N/A
LH_2 Propellant Sensors	N/A	N/A	N/A	20.0	Nil	N/A
Propellant Electronics	12.0	8.0	6.0	15.0	25.0	1
Subsystems Monitors	N/A	N/A	N/A	90.0	25.0	N/A
TOTALS				368.5	780.8	

TABLE 3.5.2-2

MANNED MISSION COMMUNICATIONS AND DATA HANDLING EQUIPMENT LIST

Subsystem Element	Dimension(in.) L	W	H	Unit Mass(Lb.)	Unit Power(watts)	Quantity Per Vehicle
Communications						
Transponder	14.3	6.0	4.8	13.9	36.0*	2
RF Power Amplifier	11.0	13.0	5.0	13.0	110.0*	2
Diplexer	7.3	2.4	3.4	2.0	Nil	2
RF Switch	2.7	4.0	1.2	0.6	Nil	2
Omni Antenna	14.6	5.6 (Cone)		1.5	Nil	4
Cabling	N/A	N/A	N/A	40.0	Nil	N/A
Data Handling						
DMU #1	16.9	8.0	6.0	34.0	127.2	1
DMU #2	28.8	8.0	6.0	64.0	216.0	1
DMU #3	25.0	8.0	6.0	45.0	201.9	1
DMU #4	18.3	8.0	6.0	40.0	156.0	1
DMU #5	18.3	8.0	6.0	44.0	100.0	1
Data Bus	N/A	N/A	N/A	40.0	Nil	N/A
Instrumentation						
LO_2 Propellant Sensors	N/A	N/A	N/A	15.0	Nil	N/A
LH_2 Propellant Sensors	N/A	N/A	N/A	20.0	Nil	N/A
Propellant Electronics	12.0	8.0	6.0	15.0	25.0	1
Subsystems Monitors	N/A	N/A	N/A	90.0	25.0	N/A
TOTALS				445.0	909.1	

*One transponder operating with receive plus transmit (36 watts) and one receive only (12 watts). Only one RF power amplifier on.

**Installed in manned module only. Not included in weight and power totals for the vehicle.

Communications. The system design approach for the cost optimum configuration incorporates two omni antennas mounted on diametrically opposed sides of the vehicle to achieve hemispherical coverage on each side. Spherical coverage is achieved by switching between the opposing antennas using a make before break latching R. F. switch. The 20-Watt Power Amplifier is an all solid-stage unit that operates on any frequency in the 2200 MHz to 2400 MHz STDN/TDRS transmit frequency range. The amplifier:

a. Amplifies the 2.5 watt R.F. output of the STDN/TDRS Transponder to a level of 20 watts without degrading the fidelity of the input signal.
b. Accepts decoded command in the form of an external switch closure for turning ON and OFF.
c. Provides status monitoring of
   1. R.F. Power (Analog; 0-20 watts).
   2. Power Supply Voltage (Analog; 0-5 VDC).
   3. Temperature (Analog; $-34^{o}C$ to $+71^{o}C$).

The STDN/TDRS Transponder function is to provide a telecommunications link between the vehicle and earth via the TDRS satellite link, or direct to earth via the STDN or to the Orbiter via a two way link. The STDN/TDRS Transponder is configured to either STDN or TDRS mode on command and consists of the following major components; receiver, spread spectrum processor, transmitter, and power converters.

In the STDN format, PCM data inputs to the transmitter are used to phase shift key (PSK) a standard SGLS 1.024 MHz subcarrier oscillator. The interrange instrumentation group (IRIG) FM subcarrier composite input from the EMU directly FM Modulates the standard SGLS 1.7 MHz subcarrier oscillator. The amplitude level is adjusted to comply with the modulation index required and linearly summed with the STDN side tone of 500KHz, 100KHz, etc., from the receiver. The ranging signal will be set for a linear 1:1 turnaround ratio for two tones. These three signals (PCM, FM, and ranging) are summed in the baseband module of the transmitter. The composite output then PM modulates the transmitter R.F. output.

For the purposes of command reception, the transponder is compatible with both the GSTDN and TDRS signal formats. Two modes are provided to (1) allow communication in the GSTDN mode only for hardline inputs, GSTDN and Orbiter and (2) dual mode operation for flight to either GSTDN or TDRS. The dual mode logic is such that either detector samples the incoming signal for power. Any signal stronger than -100dBM will switch the transponder to the GSTDN Mode and hold it there until lock is lost. Signals less than -110dBM will switch the transponder to the TDRS mode.

Hardline commands may be received in the 16 KHz PSK format at baseband through a receive side-port. To operate in this mode, an auxiliary enable must also be input from an external source such as the Orbiter or T-O umbilical. This allows command operation without R.F. radiation.

R.F. uplinked commands from GSTDN or Orbiter are received through one of the two conical log spiral antennas and the receiver portion of the transponder. Once the receiver acquires (within 1/2 second) and adequate signal presence has been established by the receiver, the receiver demodulator is activated and the command tone can be processed. Once the command bit synchronizer is locked up (providing at least a 10^{-4} BER) the decoder is deactivated to prevent noise bursts or other receiver outputs from reaching the decoder. When receiving from TDRS, the receiver searches and acquires the forward link spread spectrum signal within 20 seconds. Commands are extracted from the PN code in the despreader and handled in the command bit synchronizer in a manner similar to the GSTDN mode.

The command decoding is done partially in hardware (DMU) and partially in software. The spacecraft address will be different for each side (A) or (B), hence, only one side is allowed to pass through to the hardware decoders in the DMU and into the DMU processor. Because it is only possible for one Command channel to receive a good command, the processor conversational links will provide that command to the processor thereby effectively cross strapping the command channels via software. Each uplinked command received is subsequently telemetered in the downlink bit stream. These command replicas are then verified on the ground.

For ranging purposes the transponder provides coherent STDN side tone ranging turnaround of the uplink ranging tones. These tones are 400 KHz, 100 KHz, 20 KHz, 4 KHz, 800 Hz, 160 Hz and 40 Hz. Once the receiver is phase locked to the uplink command carrier, a phase coherent signal is sent to the transmitter oscillator, locking it coherently. Additionally, the ranging tones are detected in the receiver and sent to the transmitter baseband module where they are summed with the telemetry of 1.024 MHz subcarrier. When the TDRS Mode is selected, the ranging turnaround implementation is such that the spread spectrum PN return link code will be synchronized to the forward link PN code (Data group 1, mode 1) providing two way range and doppler measurements.

Data Handling. The data management unit (DMU) provides the interface for commands and measurements between its internal processors and other OTV subsystems. The DMU samples analog, discrete (bilevel), and serial measurement data, formats the data into serial digital words and provides the formatted words to the transponder. The DMU performs redundancy management by processing internal bite status discretes, selecting which side is in control. Each DMU contains its own internal processor.

The DMU accepts analog inputs, conditions as required, and converts each 0 to 5 volt signal sampled into a 8-bit binary coded byte.

The types of command output functions provided by the DMU are:

a. Antenna Relay Switching. The DMU provides the capability to drive make coils and break coils for switching between pulses, with the break pulse lagging behind the make pulse, to provide make-before-break antenna switching.

b. TVC Commands. Two digital-to-analog converters are implemented in each DMU to convert yaw and pitch magnitude commands (generated by the processors) to analog voltages to drive the TVC actuators.

c. RCS Drivers. Logic is provided to decode and turn ON/OFF the appropriate thruster valves based on commands transmitted from the side in control.

d. Relay Matrix. The DMU has the capability of switching the relays in the relay matrix upon command from its internal processor.

e. Spacecraft Discrete Commands. Eight solid state drivers are appropriate logic to decode and turn ON/OFF selected spacecraft discretes based on commands transmitted from the side in control.

Distributed processing is used in the design concept with the processing function being related to the equipment/subsystems connected to the DMU. As shown previously in Figure 3.5.2-1, DMU #1 accomplishes communications control and monitor and telemetry formatting. Vehicle GN&C and mission sequencing is accomplished by DMU #2. Electrical power system control and monitor is accomplished by DMU #3. And DMU #4 performs all main propulsion system control and monitor. Each DMU contains redundant processors for the cost optimum configuration. To accommodate the manned mission, an additional processor and power supply is installed in space provided within the DMU to bring the configuration up to triple redundancy of processors and power supplier.

3.6 ELECTRICAL POWER SUBSYSTEM

The electrical power supply and distribution subsystem provides (1) power for vehicle equipment and the spacecraft, (2) switching and distribution of electrical power to the vehicle and spacecraft, (3) interconnecting cabling for the avionics and other subsystems, and (4) the capability to use power supplied by the Orbiter, ground support equipment, or the Space Station when attached.

3.6.1 Top Level Requirements

The top level requirements for the electrical power subsystems are:

a. Provide power to all vehicle subsystems.
b. Provide capability to supply power to vehicle subsystems from the ground or Orbiter when in the launch configuration, from internal sources when deployed, and from the Space Station when attached.
c. Provide redundancy of internal power sources.
d. Control and distribute power to all vehicle subsystems.
e. Provide interconnecting wiring for all vehicle subsystems except for instrumentation wiring and RF cabling.
f. Provide 200 watts of power to a payload when attached.

The energy requirements for a nominal GEO delivery mission are summarized in table 3.6.1-1. The maximum power usage, during main engine burns, is 1859 watts. The power usage during coast is 1332 watts.

3.6.2 Hardware Definition

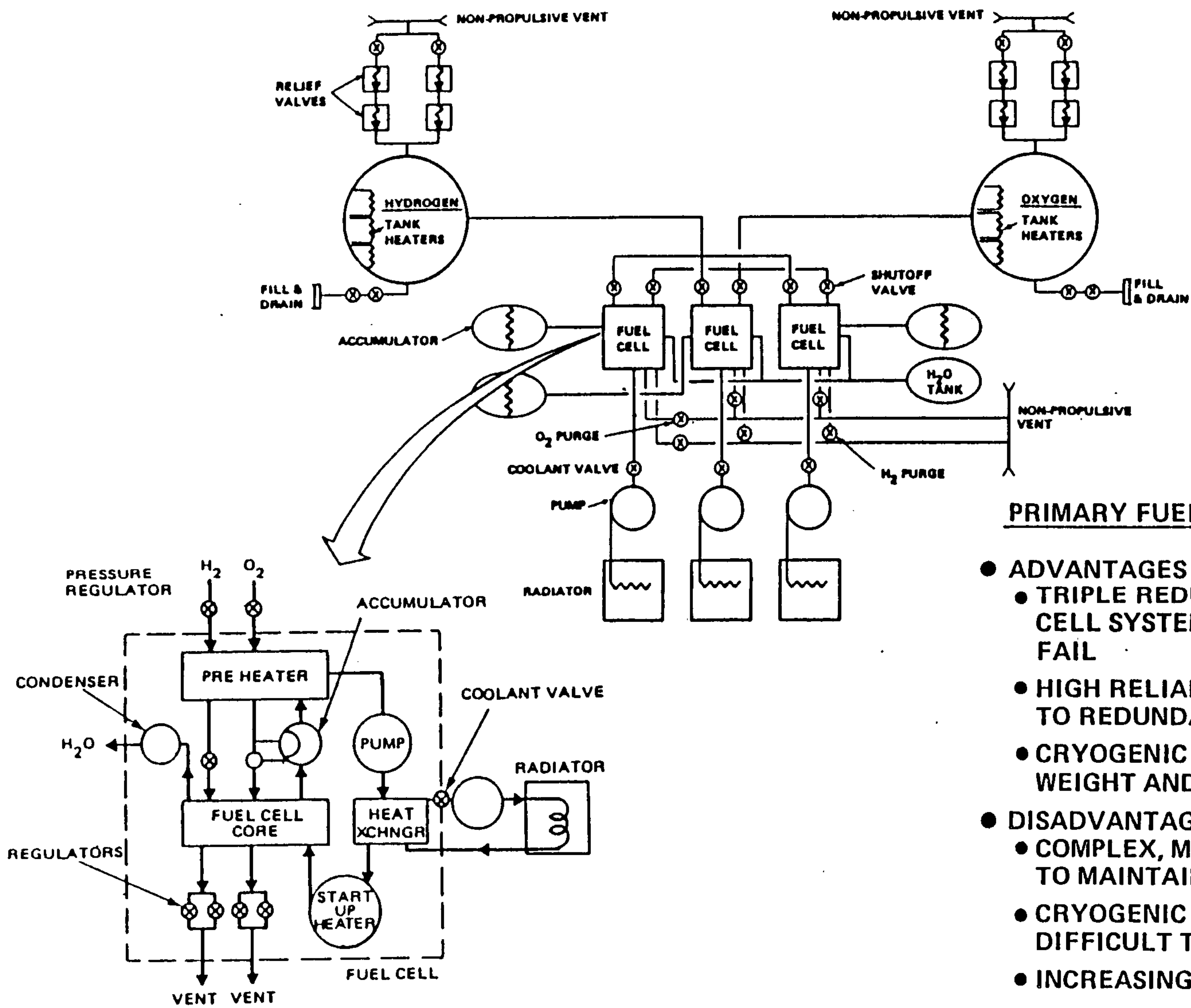
The primary power sources for the electrical power subsystem are 28 VDC, 2 kilowatt hydrogen/oxygen fuel cells. The schematic for the fuel cell system for the manned mission configuration is shown in figure 3.6.2-1. As discussed in section 10.0 (Reliability/Safety) of Volume II, Book 3 (Vehicle Level Analysis), an analysis was performed to determine the most cost effective approach to vehicle subsystem configuration for the OTV program considering both the unmanned and manned missions. The results of the analysis showed that it was more cost effective to configure the electrical power subsystem initially with two fuel cells for the unmanned mission. Included in this cost optimum configuration was the equipment required to accommodate an additional fuel cell and its control for the manned missions.

Table 3.6.2-1 presents a listing of the electrical power subsystem equipment for the cost optimum configuration. For the manned mission configuration an additional fuel

Table 3.6.1-1. Reference Power Requirements (51.2 Hour GEO Delivery Mission)

Subsystem	Power (Watts)	Time Duration (Hours)	Energy Usage (Watt-Hours)
Avionics			
Guidance and Navigation	215	51.2	11,008
Communications	146	51.2	7,475
Data Handling	585	51.2	29,952
Instrumentation - Eng Burn	50	0.5	25
- Coast	25	49.7	1,243
Propulsion			
Engine - Coast	30	49.7	1,491
- Burn	168	0.5	84
TVC Actuators	560	0.5	280
Attitude Control			
Thrusters	1.4	51.2	72
Heaters	36	51.2	1,843
Power			
Distribution Losses[1]			
- Eng Burn	88	0.5	44
- Coast	52	49.7	2,584
	10	36.0	360
Heaters	32	51.2	1,638
Payload	200	36.0	7,200
Emergency	100	24.0	2,400
	Total Energy Usage		= 67,699

[1] 5% of load



PRIMARY FUEL CELL—2kW

- ADVANTAGES
  - TRIPLE REDUNDANT FUEL CELL SYSTEM—2 OF 3 MAY FAIL
  - HIGH RELIABILITY DUE TO REDUNDANCY
  - CRYOGENIC TANKS—LOW WEIGHT AND VOLUME
- DISADVANTAGES
  - COMPLEX, MORE DIFFICULT TO MAINTAIN
  - CRYOGENIC TANKS—MORE DIFFICULT TO FILL
  - INCREASING COST

Figure 3.6.2-1 Baseline Power System

OTV-1573

Table 3.6.2-1. Electrical Power Subsystem Equipment List for Cost Optimum Configuration

Subsystem Element	Dimensions (in.) L	W	H	Unit Mass(lb)	Quantity/ Vehicle
Fuel Cell System					
Fuel Cell (2kW)		TBD		80	2
O_2 Tank Assembly		15.6 dia		23	1
H_2 Tank Assembly		19.7 dia		26	1
Relief Valve		TBD		0.5	4
Solenoid Valve		TBD		0.5	2
Disconnect		TBD		0.5	2
Accumulator		TBD		3	2
Water Tank		TBD		7	1
Heaters		TBD		0.5	2
Pump		TBD		4	2
Lines/Fittings	N/A	N/A	N/A	10	N/A
Heat Exchanger		TBD		5	1
Coolant Plumbing	N/A	N/A	N/A	10	N/A
Pump		TBD		5	1
Radiator		TBD		49	N/A
Coolant	N/A	N/A	N/A	5	N/A
Battery (Ni/H)		TBD		25	1
Distribution and Control					
Power Distribution Unit	18.0	12.0	11.0	40	2
Energy Control Unit	14.1	9.0	10.2	31	1
Power Transfer Unit	15.3	8.2	11.4	20	1
Wire Harness	N/A	N/A	N/A	180	N/A
Reactants					
Usable	N/A	N/A	N/A	83	N/A
Residuals	N/A	N/A	N/A	11	N/A
		Total Dry Weight =		655	
		Total Weight =		749	

cell (with the necessary plumbing, valves, and control) is added. The fuel cell size (2 kW) was selected to provide the capability for a single cell to be able to supply the total mission power loads. Thus, for the manned mission, triple redundancy is provided. Supplemental power is provided from a battery source to provide smoothing from transients during valve switching. The nickel/hydrogen battery design is rated at 25 ampere hours. The equipment list for the manned mission configuration is shown in table 3.6.2-2.

The OTV distribution subsystem consists of power distribution units, power transfer unit, wiring harnesses, and umbilical connectors. The power distribution units interface with other vehicle subsystems and the ASE and, in addition, provide for electrical power transfer switching between external and internal power sources. Separate power busses are provided for redundant subsystems. Measurements of bus currents and voltages are provided to the data management unit (DMU) for entry into the telemetered data stream.

The power distribution units provide relay switching functions required for control of discrete vehicle elements and power switching such components as heaters, transponder transmitters, power amplifiers, star tracker, and propellant measurement electronics. These relay switching functions are controlled by the data management units. The power transfer unit provides for switching the spacecraft power input between external and internal vehicle power sources. The distribution system was shown in figure 3.5.2-1 of this book.

Connectors, cabling, and umbilical connectors are provided for connections between OTV subsystems, between the OTV and the ASE, and between the spacecraft and the ASE and OTV subsystems. Not included in the distribution subsystem are instrumentation wiring between sensors and the SCU and RF cabling within the communications subsystem.

3.6.3 Functional Description

The power source and distribution subsystem provides nominal 28 VDC power to the vehicle subsystems and also provides 200 watts to the payload. The system provides redundant power busses to supply mission critical hardware.

Fuel Cell System. The characteristics used in defining the weight and reactant consumption are those for the modified Shuttle Orbiter hardware. The cell area is 0.25 square feet with the control/plumbing section of the cell assembly tailored to the reduced (from the Shuttle Orbiter cell) cell area. During normal mission operations both

Table 3.6.2-2. Electrical Power Subsystem Equipment List for Manned Mission Configuration

Subsystem Element	Dimensions (in) L	W	H	Unit Mass(lb)	Quantity/ Vehicle
Fuel Cell System					
Fuel Cell (2KW)		TBD		80	3
O_2 Tank Assembly		15.61 dia		23	1
H_2 Tank Assembly		19.71 dia		26	1
Relief Valve		TBD		0.5	8
Solenoid Valve		TBD		0.5	4
Disconnect		TBD		0.5	2
Accumulator		TBD		3	2
Water Tank		TBD		7	1
Heaters		TBD		0.5	2
Pump		TBD		4	3
Lines/Fittings	N/A	N/A	N/A	10	N/A
Heat Exchanger		TBD		9	2
Coolant Plumbing	N/A	N/A	N/A	10	N/A
Pump		TBD		10	2
Radiator		TBD		52	1
Coolant	N/A	N/A	N/A	6	N/A
Battery (Ni/H)		TBD		50	2
Distribution and Control					
Power Distribution Unit	18.0	12.0	11.0	40	2
Energy Control Unit	14.1	9.0	10.2	31	1
Power Transfer Unit	15.3	8.2	11.4	20	2
Wire Harness	N/A	N/A	N/A	200	N/A
Reactants*					
Usable	N/A	N/A	N/A	37	N/A
Residuals	N/A	N/A	N/A	5	N/A
		TOTAL DRY WEIGHT =		811 LBS	
		TOTAL WEIGHT =		853 LBS	

* Reactant Weights Reflect Use Of MGSS Power Except During Active Vehicle Transfer Periods.

Table 3.6.2-2. Electrical Power Subsystem Equipment List for Manned Mission Configuration

fuel cells are operating. However, either of the fuel cells will sustain the nominal mission load in the event of a failure. For the manned mission, the third cell provides triple redundancy.

The fuel cell H_2 and O_2 reactant tanks supply the reactants on demand at an oxidizer to fuel ratio of 8 to 1 to meet the power demand at a nominal use rate of 0.85 lb/kW-hr. The reactants are maintained at supercritical pressures of 250 psia in the hydrogen tanks and 900 psia in the oxygen tank during use by means of tank pressure controlled electric heaters. Reactant pressures are reduced to the fuel cell operating pressure of approximately 16 psia by the fuel cell inlet demand regulators.

Distribution. The primary interface between vehicle subsystems and the vehicle power source are the power distribution units. The normal mode switching of command outputs on the OTV is accomplished by command outputs from the Data Management Unit (DMU). Those command loads that require high currents and have high inductive reactances are interfaced with the DMU through the power distribution unit. In addition, provisions are incorporated into the power distribution unit to accommodate hardwire commands from the ground via the ASE, from the Space Station, or from the ASE and hardware switch, and for bus status indications.

The power transfer unit (PTU) provides motor driven switch capability for switching of the spacecraft power source between the external and OTV power sources. Switching commands are supplied by and switch status is provided to the power distribution unit.

4.0 LAUNCH PROCESSING OPERATIONS

This section summarizes the major operations required to prepare the GBOTV for each flight. A more indepth discussion is found in Volume II Book 4.

4.1 OTV MAIN STAGE WITH SPACECRAFT

4.1.1 Timelines

The top level operations timeline for the case of the GBOTV and a spacecraft (payload) being launched together is shown in Figure 4.1-1. Shuttle related timebars are based on STAR 27, Figure 8, Level III STS Turnaround Assessment and VSTAR 10, Figure 16, Level III Assessed Timeline. The refurbishment timebar is an estimate based on the configuration and characteristics of the GBOTV known at this time. The Integration and PCR timebars are derived from past IUS/spacecraft processing experience. Processing of a given OTV is estimated to last approximately 8 weeks using primarily two shift operations. To satisfy the low model OTV flight rate of 12 per year (4½ week flight centers), two parallel processing lines are used.

Additional detail on the top level refurbishment timebar is shown in Figure 4.1-2. The key task involved in the refurbishment operations includes: a) safing the propulsion and reaction control systems, b) inspection and maintenance of the main propulsion system, c) functional check of the Avionics subsystems, d) servicing of the storables, and e) installation of a new ballute.

The most significant timebar in the refurbishment operations is the "Maintain and Service Engines and Main Propulsion System." This timebar of 40 hours is somewhat arbitrary. It allows for trouble shooting the subsystem (8 hours), changeout of a complete engine (16 hours), a subsystem checkout (8 hours) and servicing (8 hours). This scenario merely substantiates the 40 hour timebar and is but one of many possible maintenance scenarios. All other refurbishment and maintenance activities except for servicing the hypergol system are accomplished in parallel with and within the same timebar. A more rigorous analysis should be accomplished after knowledge of the actual OTV hardware is available.

A breakdown of the major tasks associated with GBOTV/spacecraft integration timeline is presented in Figure 4.1-3. Further detail on the operations timeline dealing with OTV/Spacecraft integration with the STS Orbiter is shown in Figure 4.1-4.

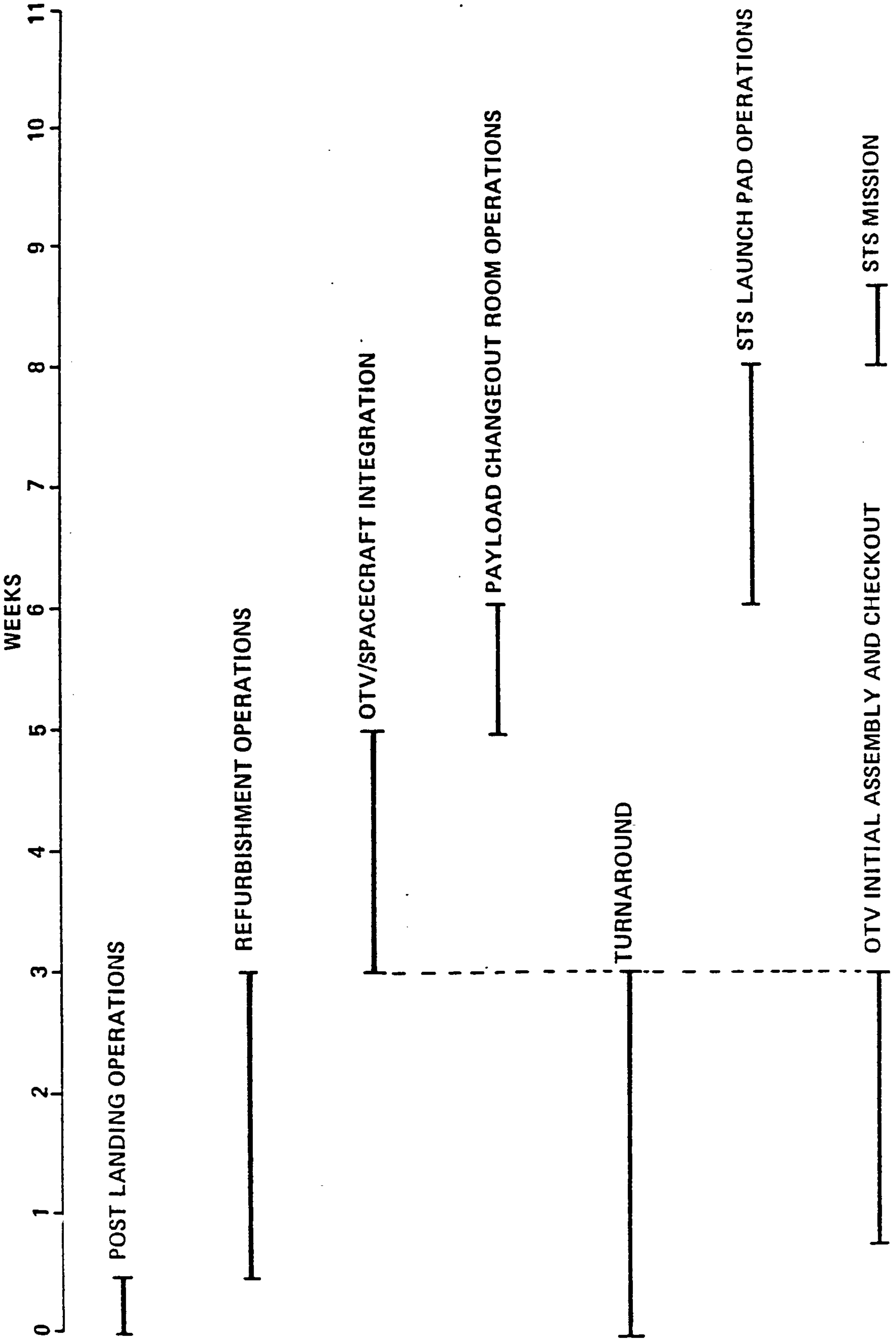
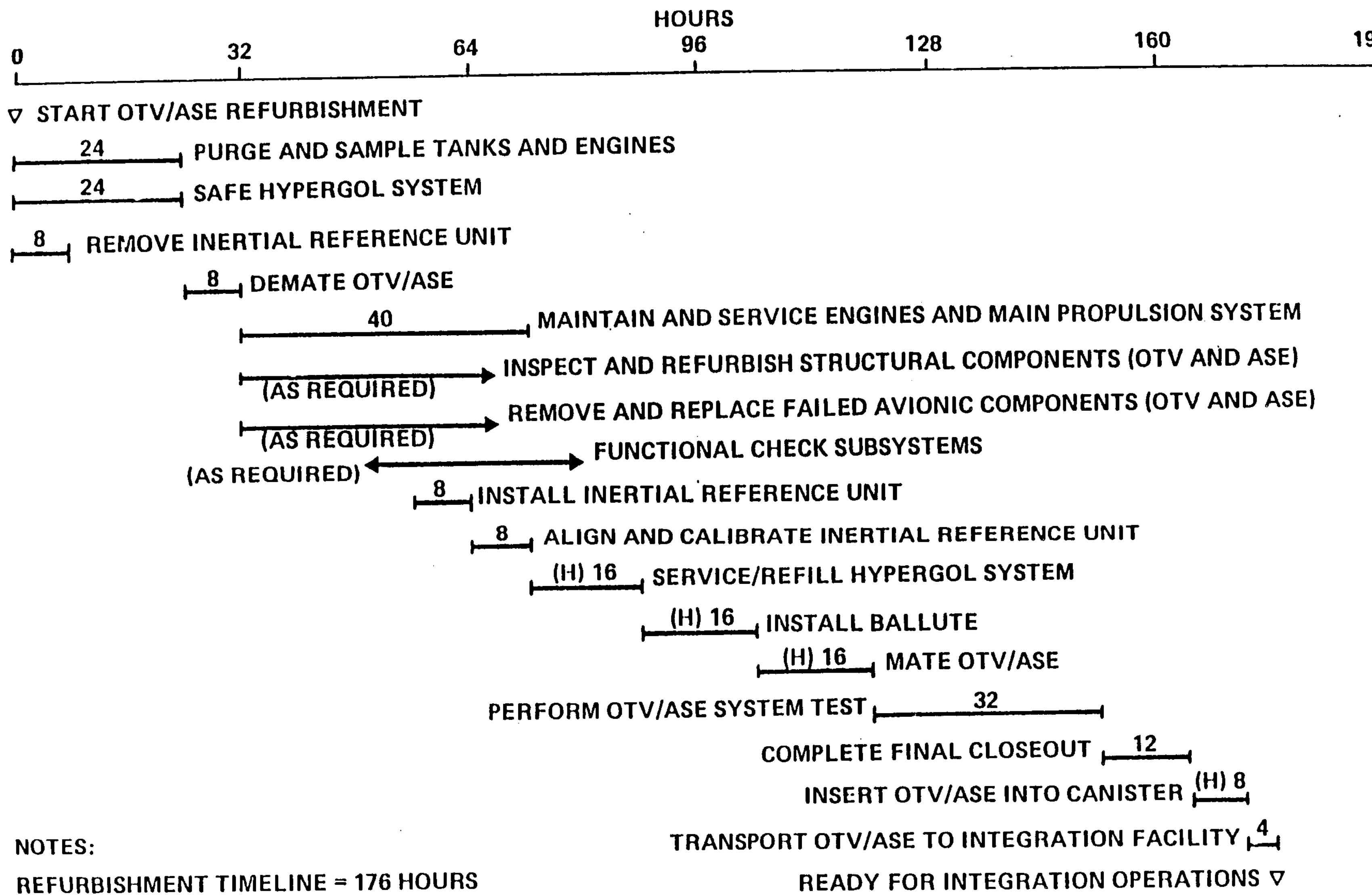


Figure 4.1-1 Ground Based OTV Operations Timeline

OTV-813



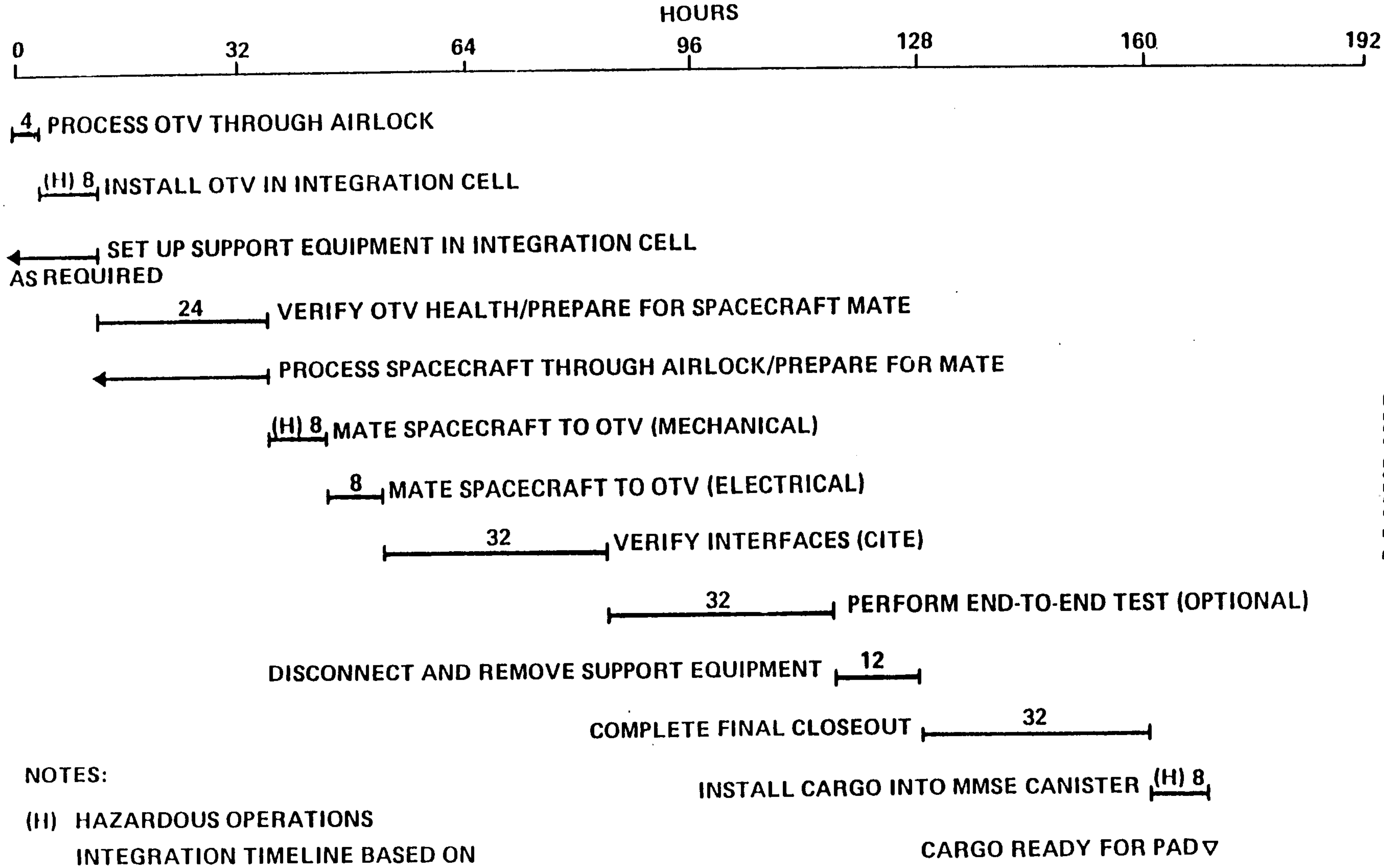
NOTES:

REFURBISHMENT TIMELINE = 176 HOURS

(H) HAZARDOUS OPERATIONS

Figure 4.1-2 GBOTV Refurbishment Operations

OTV-507



NOTES:

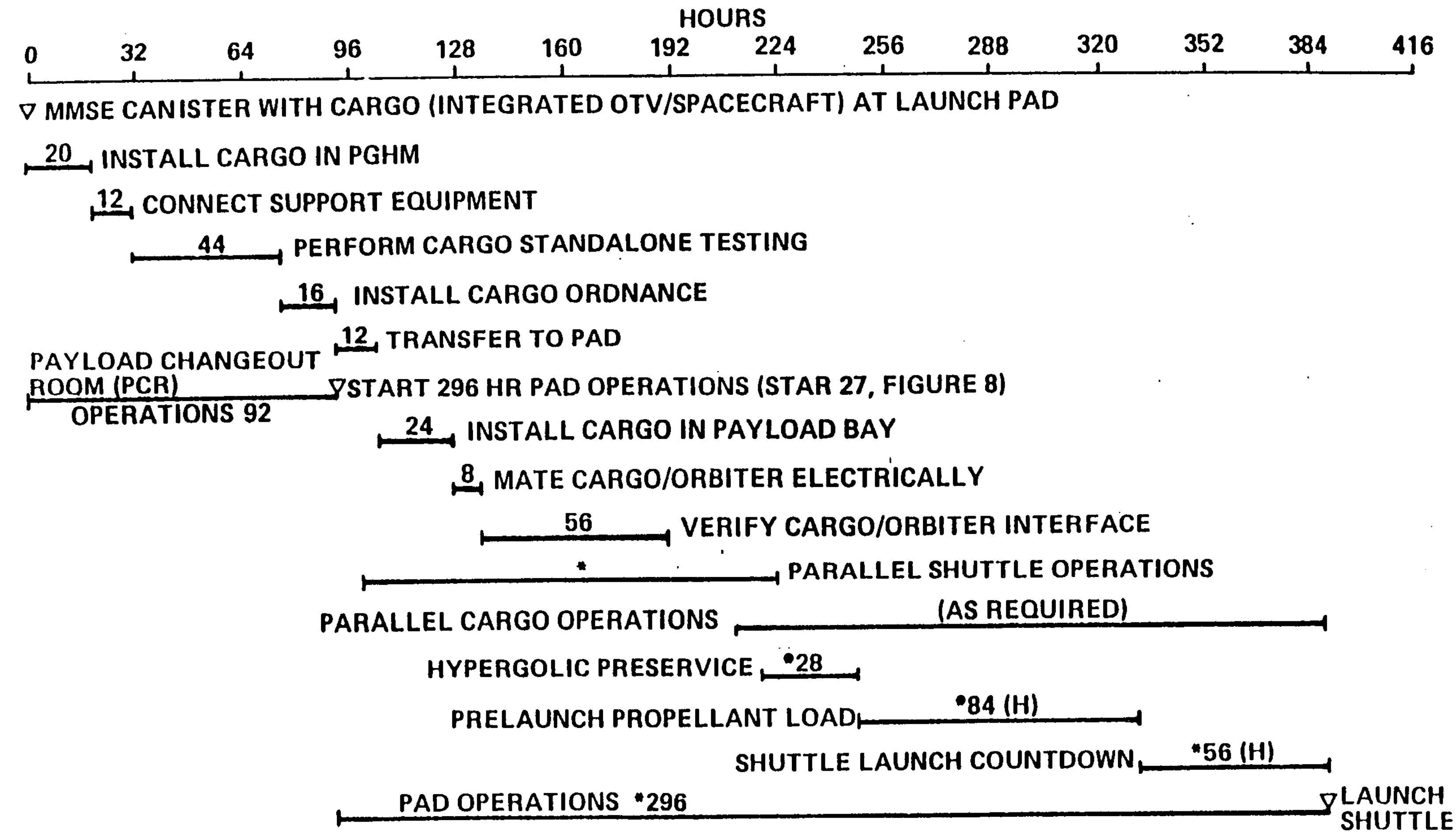
(H) HAZARDOUS OPERATIONS

INTEGRATION TIMELINE BASED ON IUS/TDRS VPF OPERATIONS

OTV/SPACECRAFT INTEGRATION TIMELINE = 168 HOURS

Figure 4.1-3 GBOTV/Spacecraft Integration

OTV-523



PCR TIMELINE (OFFLINE) = 92 HOURS

CARGO OPERATIONS TIMELINE (ON LINE) = 88 HOURS

*STAR 27, FIGURE 8, LEVEL III STS TURNAROUND ASSESSMENT

NOTES:

(H) HAZARDOUS OPERATIONS

Figure 4.1-4 OTV/Spacecraft/Orbiter Integration

OTV-503

4.1.2 Processing Effort and Organization

The processing effort expressed in terms of calendar time is shown in Table 4.1-1. The calendar time of 7.68 weeks per flight supports the maximum requirement of 12 OTV missions per year with two OTV processing lines.

The organizations and headcount to support the two processing lines on a two shift basis is shown in Table 4.1-2. The data is based on a launch site support organization developed during a previous OTV Concept Definition Study. (Document D180-26090-2, Final Report OTV Concept Definition Study, Volume 2, Mission Analysis and Operations, 1980.) The previously developed organization was modified, primarily by increasing the numbers of engineers, technicians, planners and inspectors necessary to support two shift, two line operations resulting in a 92 person organization.

The processing effort required after an OTV flight is estimated to require 10.5 man years. The methodology used to arrive at per flight costs is as follows:

a. The general tasks are assigned a work schedule. Post Landing and STS Launch Pad Operations are on a 7 day/12 hour schedule. All other operations are on 5 day/8 hour schedule. All operations are worked on a two shift basis.

b. The timeline hours are converted to calendar weeks either by dividing by 168 for the 7/12 schedule or 80 for the 5/8 schedule.

c. The calendar weeks associated with the 7/12 schedule are modified by a factor of 2.65 to account for overtime.

$$\frac{40 + 44(1.5)}{40} = 2.65$$

d. The year is assumed to have 50 weeks (vacation, holidays, and roundoff).

e. The 92 person organization supports each processing line equally (divide by 2).

f. The equivalent calendar work weeks are summed, divided by 50 and multiplied by 92/2 to arrive at the manpower requirement.

4.2 AUXILIARY PROPELLANT TANK WITH SPACECRAFT

This scenario occurs when the OTV plus auxiliary propellant tank (APT) and payload exceed STS limits. Accordingly, the APT and payload are launched together but separate from the main stage. This approach requires an OTV-to-APT/Spacecraft mating on-orbit. The tank module interfaces with the OTV at the normal OTV-to-Spacecraft interface and provides the appropriate "flow-through" plumbing, data circuits and electrical circuits. The APT interfaces with the spacecraft in a manner identified to the OTV. ASE is required to support the filled APT plus the spacecraft and to provide the Orbiter Bay to Cargo interfaces.

Table 4.1-1 Ground Based OTV Ground Processing Effort

MAJOR ACTIVITY	WORK SCHEDULE ▷1	NUMBER OF SHIFTS	PROCESSING TIMELINE HOURS (SERIAL)	CALENDAR WEEKS PER FLIGHT
INITIAL ASSEMBLY AND CHECKOUT	5 x 8	2	180	2.25
← TURNAROUND →				
POST LANDING OPERATIONS	7 x 12	2	72	0.43**
REFURBISHMENT OPERATIONS	5 x 8	2	176	2.20
OTV TANK/SPACECRAFT INTEGRATION	5 x 8	2	168	2.10
PAYLOAD CHANGEOUT ROOM OPERATIONS	5 x 8	2	92	1.15
STS LAUNCH PAD OPERATIONS	7 x 12	2	296	1.8*
TURNAROUND	---	---	TOTAL	7.68

*EQUIVALENT 5 x 8 TIME EQUALS 4.77 CALENDAR WEEKS

**EQUIVALENT 5 x 8 TIME EQUALS 1.14 CALENDAR WEEKS

▷1 DAYS PER WEEK X HOURS PER DAY

OTV-1746

Table 4.1-2 Ground Based OTV Ground Processing Organization

SKILL CLASSIFICATION	HEADCOUNT	PERCENTAGE OF TOTAL	HEADCOUNT BREAKDOWN
ADMINISTRATION (CONTRACTS, FINANCE, PERSONNEL, PROCUREMENT)	3	3.3	CONTRACT ADMINISTRATION (1), FINANCE/PERSONNEL (1), PROCUREMENT (1)
DATA PROCESSING (COMPUTER SPECIALISTS, DATA ENTRY)	1	1.1	COMPUTER TECHNICIAN/ PROCESSOR (1)
ENGINEERING (ENGINEERS, DRAFTING, ASSOCIATE ENGINEERS)	27	29.3	TEST AND CHECKOUT (2x2x6=24) SUPPORT (3)
LOGISTICS (STOREKEEPERS, SHIPPING AND RECEIVING, DRIVERS)	4	4.3	SHIPPING AND RECEIVING (1), STOREKEEPER (2), DRIVER (1)
MANAGEMENT (MANAGEMENT AND SUPERVISION)	4	4.3	MANAGER (1), RECEPTIONIST (1), SUPERVISOR (2)
OPERATIONS PLANNING (PLANNERS, SCHEDULERS, ANALYZERS, DOCUMENTATION)	7	7.6	PP&C (1), PLANNERS (4), ANALYST (2)
QUALITY/INSPECTOR (CONFIGURATION MANAGEMENT, QUALITY ASSURANCE)	10	10.9	TEST AND CHECKOUT (2x2x2=8) CONFIGURATION CONTROL (1) LOGISTICS INSPECTOR (1)
TECHNICIANS (MECHANICAL AND ELECTRICAL TECHNICIANS, EQUIPMENT OPERATORS, FUELS SPECIALISTS)	36	39.1	TEST AND CHECKOUT (2x2x9=36)
TOTAL	92	99.9	

OTV-588

Figure 4.2-1 depicts the processing required to turnaround the APT after the first flight. The APT returns to earth as part of the GBOTV. It is demated from the OTV, refurbished and mated to it's ASE. The turnaround timeline includes a subsequent spacecraft integration and STS launch. A total of 6.8 calendar weeks are required to turnaround the reusable APT (from landing to relaunch).

4.3 ON-ORBIT PROCESSING

The operational timeline for the mating of a auxiliary propellant tank/spacecraft with the OTV main stage is shown in figure 4.3-1. The main stage is transported in the second STS flight so as to minimize cryogenic fuel boiloff. The APT/Spacecraft is stored at the Space Station until the GBOTV arrives. Based on two people being present during EVA operations a total of 9.6 EVA crew hours and 6.8 IVA crew hours have been estimated as necessary to perform the physical integration and checkout of the OTV elements.

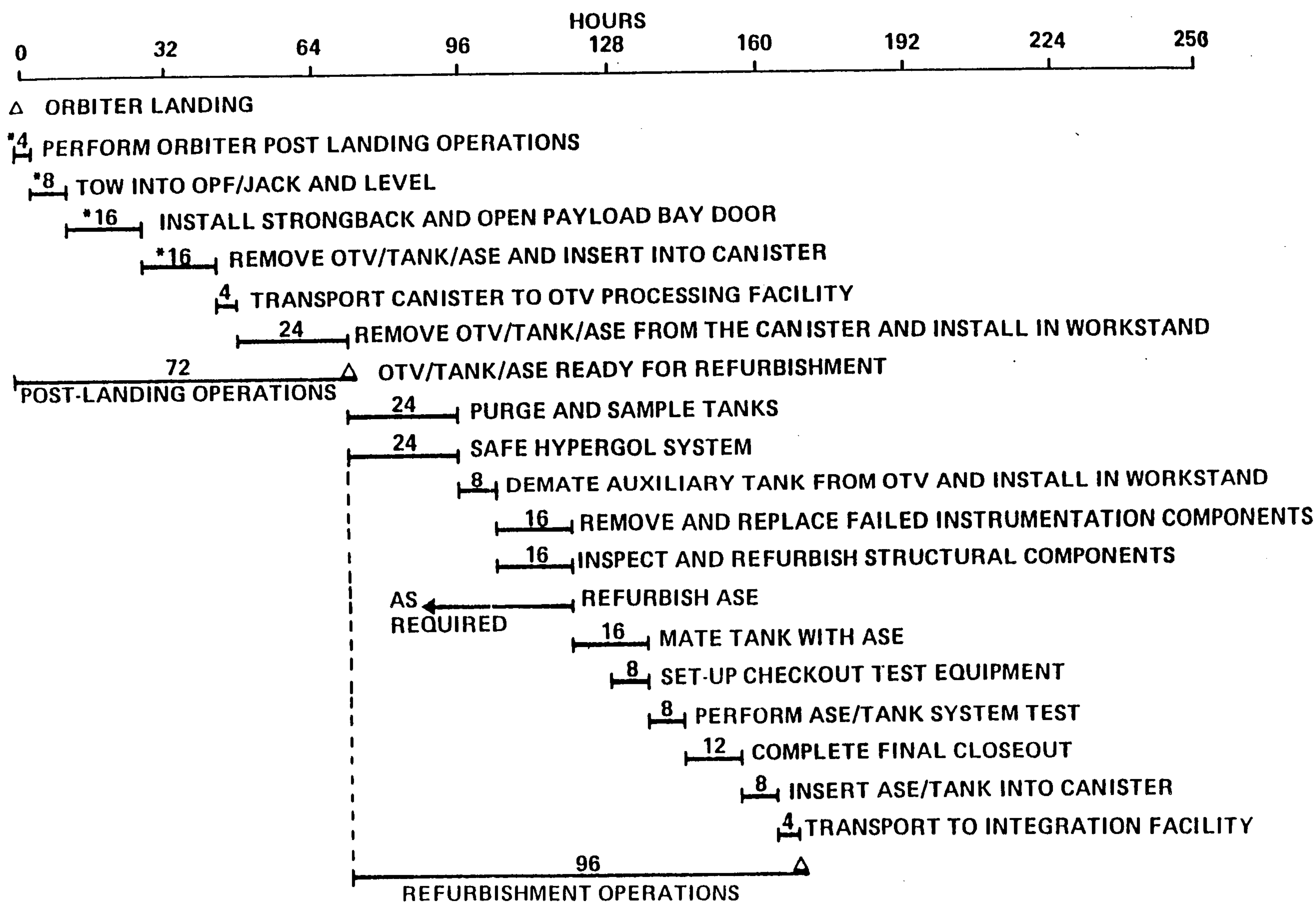


Figure 4.2-1 Reusable Auxilliary Tank Turnaround Timeline

OTV-628

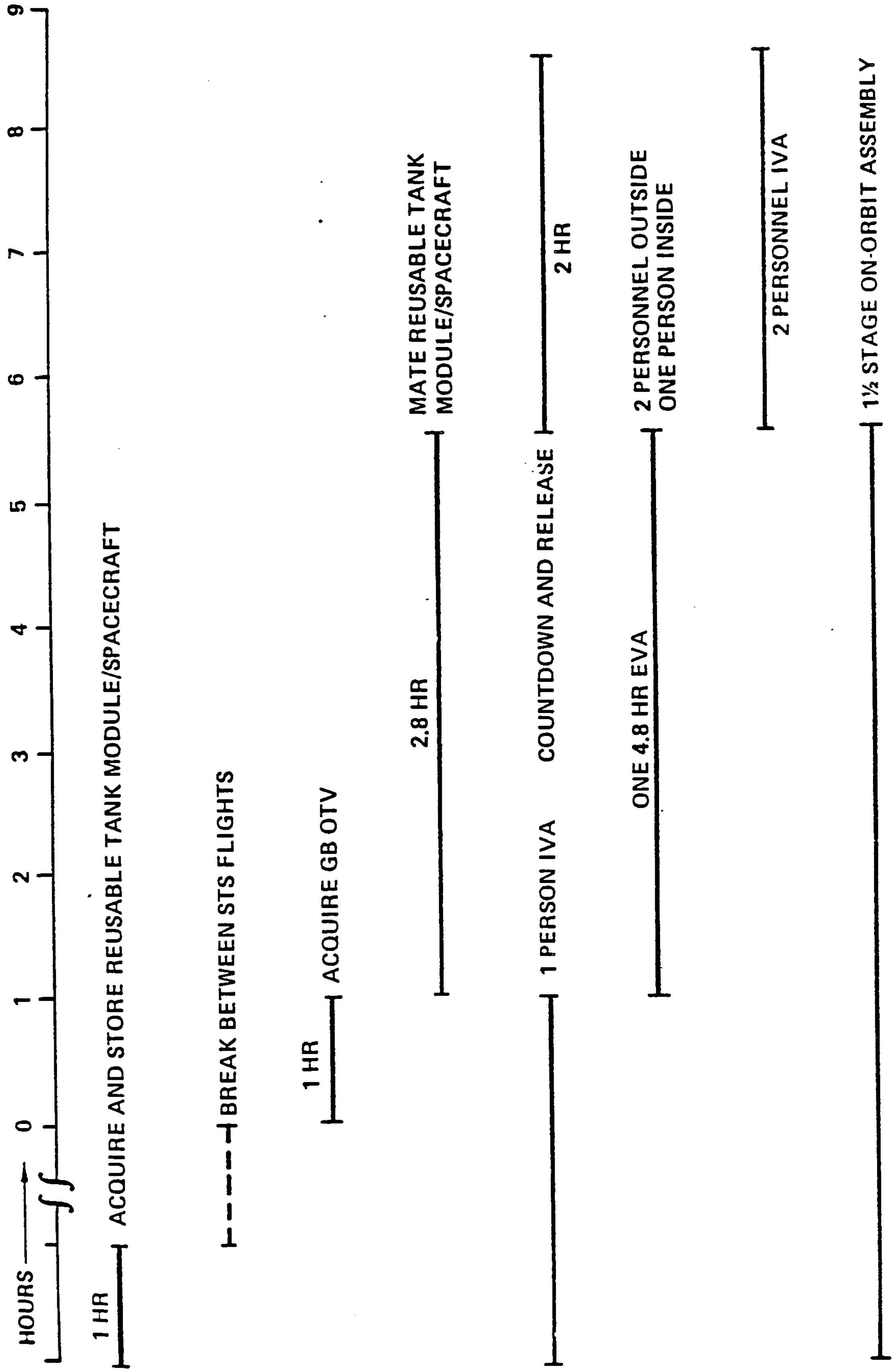


Figure 4.3-1 GB OTV/Space Station Operations

5.0 PROGRAMMATICS

This section will focus on the top level features of the recommended OTV program in terms of system utilization, program schedule and cost.

5.1 SYSTEM ELEMENTS AND UTILIZATION

The time phased requirements for the major elements associated with the GB OTV program are shown in Figure 5.1-1. The main stage with 47 k-lbs propellant is adequate until 2001. At that time, an auxiliary propellant tank is added for the 20 k-lbs payloads and manned sorties with the latter requiring the maximum loading at 78 k-lbs. Three sizes of ballutes are included in the inventory to allow match up with the three reentry conditions: stage only, stage + auxiliary tank, stage + auxiliary tank + manned module. Redundancy provisions primarily reflect a cost optimum complement of equipment for unmanned missions. Additional components are added for each of three manned flights and removed after each flight to minimize the performance impact. Finally, modest OTV accommodations need to be incorporated at the Space Station at the time when an auxiliary propellant tank is required.

5.2 PROGRAM SCHEDULE

The overall program schedule for the ground-based OTV is shown in Figure 5.2-1. A more indepth stage development schedule is presented in Figure 5.2-2. The development is accomplished in three phases, beginning initially with the basic stage and the 33 ft ballute. When the mission requirements increase, the auxiliary tanks and larger ballutes are developed. The pacing item for the program involves having data available from an aeroassist flight experiment (AFE) using a ballute before the stage PDR occurs. The goal would be to have the data at least 2 - 3 months before the PDR so appropriate assessment can be made and contribute to the detail design effort leading up to stage CDR. To accommodate a 1994 IOC GB OTV using a ballute, the ballue AFE test would have to occur in late 1989 and the initial AFE test involving a shaped brake would occur in early 1989. As of this writing, the first AFE test now appears to be scheduled for no sooner than late 1991 resulting in the ballute AFE test being in 1992. Accordingly, the reusable GB OTV using a ballute would be delayed until late 1996 or early 1997 if the same relationship between the AFE test and strage PDR is maintained.

Higher risk schedules could be developed however that would allow the stage to begin operation in 1994. One approach would be to rely on analytical and ground test facilities. The tests would use wind tunnels for aerodynamics and control characteristics, water tanks for physical properties and control characteristics, and

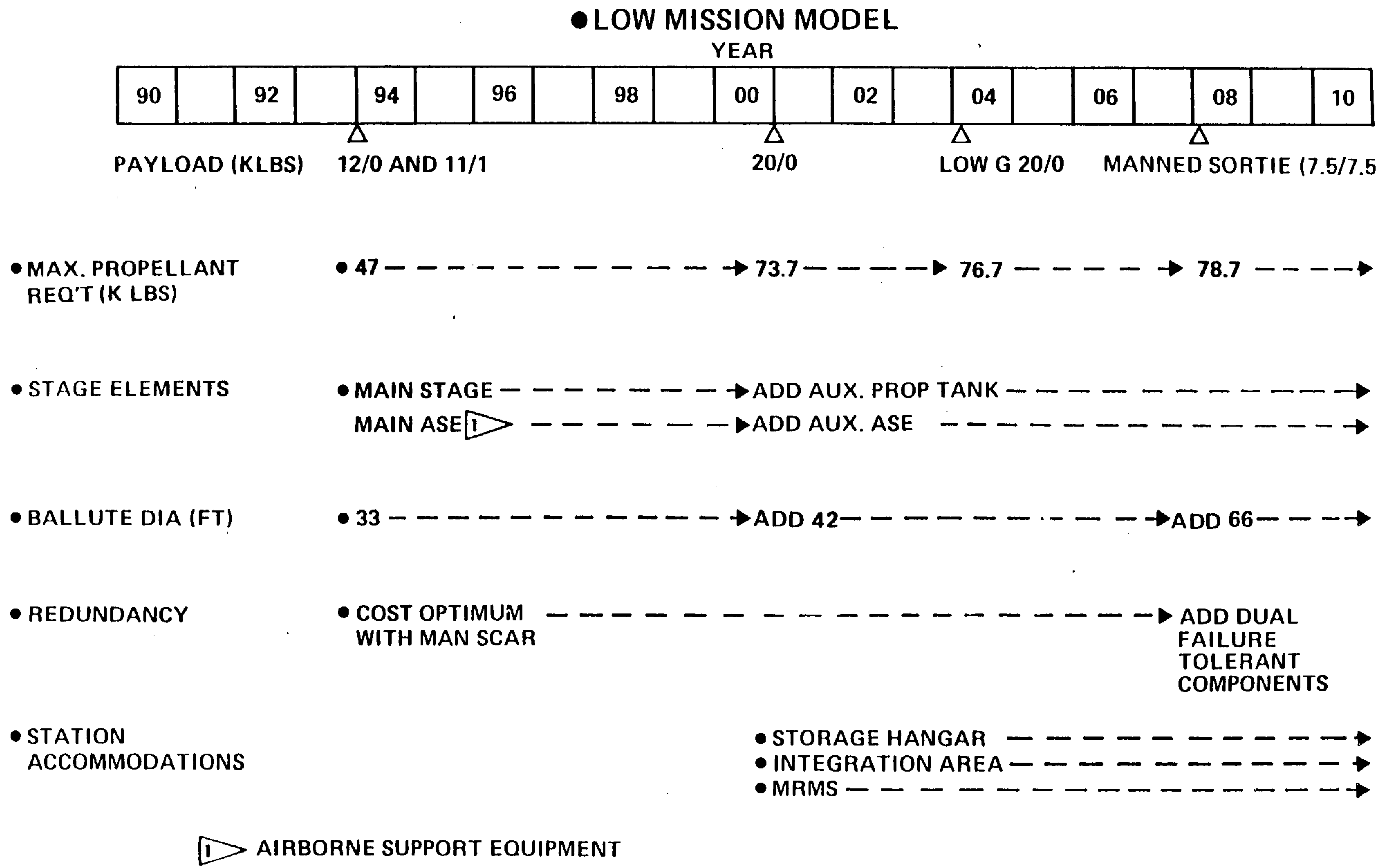


Figure 5.1-1 Time Phased System Requirements — Ground Based OTV Program

OTV-904

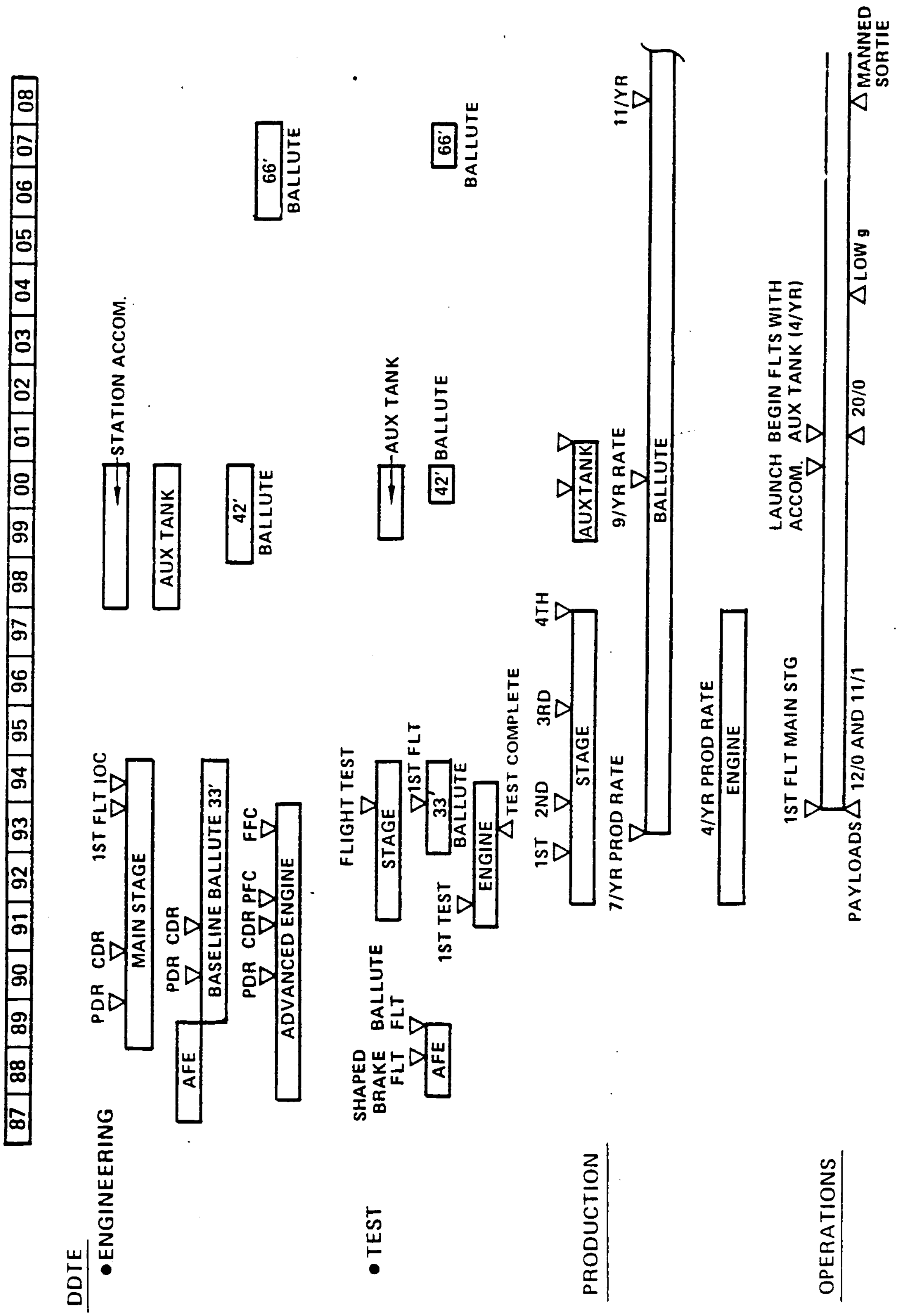
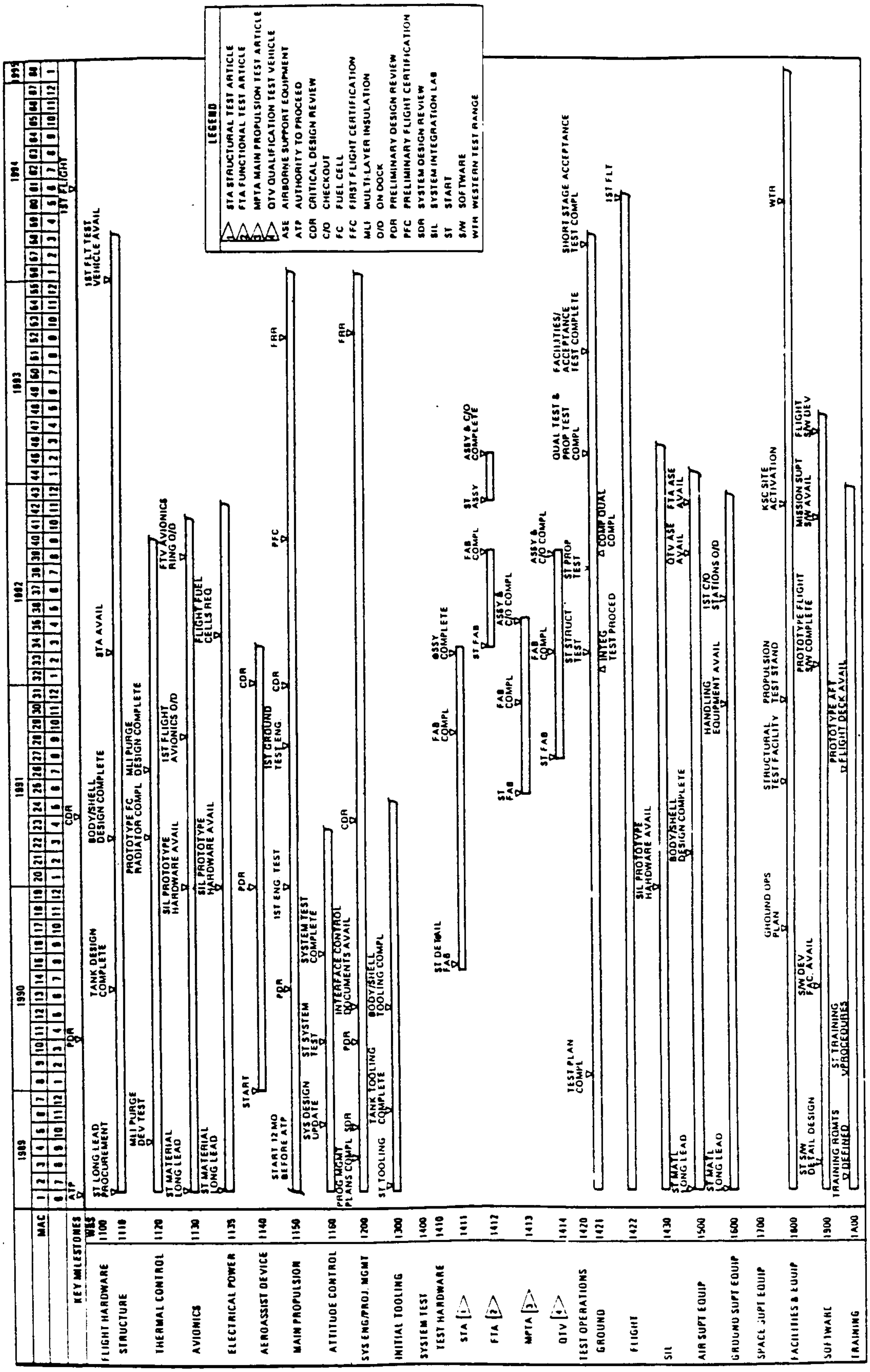


Figure 5.2-1 Proposed Program Schedule — Ground Base OTV



OTV-963

Figure 5.2-2 OTV Development Schedule – Preliminary thermal facilities to establish thermal protection properties. Another option would maintain the AFE test program according to more recent estimates of when the test would occur and have the stage developed for a 1994 flight but be flown all propulsively and expendable until the AFE data is available. Identification of a preferred approach will not be made at this time because more data is necessary regarding the schedule of all space transporation related programs. The pacing item for the stage DDT&E phase is the advanced engine development covering a 6 year period.

The stage DDT&E test program which includes a complete ground test program and one flight test. The ground test objectives and approach are shown in Table 5.2-1. Major test hardware includes a structural test article (STA), facilities test article (FTA), main propulsion test article (MPTA) and a systems integration laboratory (SIL) unit. Qualification testing is accomplished with the first flight vehicle. Four stages and two auxiliary tanks satisfy the production needs for the low mission model. The ballute production rate corresponds with the vehicle flight rate. All main engines are produced in five years in order to achieve a reasonable production rate.

5.3 PROGRAM COST

The total program funding is shown in Figure 5.3-1. The DDT&D portion of the program is $1317 million and has a peak funding of $250 million. A breakdown of the DDT&E is shown in Figure 5.3-2. Major items within DDT&E include the related technology program ($155 million for main engine and a dedicated aeroassist flight experiment) the main stage ($962 million) auxiliary propellant tank ($144 million) and station accommodations ($56 million). The most significant items within the main stage are the main engine ($300 million) and software ($110 million). The DDT&E also includes the cost associated with the first flight units. The total production cost is estimated at $224 million. A breakdown of the production cost is presented in Figure 5.3-3. Operations cost averages about $950 million per year. The cost per flight is shown in Table 5.3-1. Use of the main stage only is estimated to be $79 million and $143 million for flights involving both the mainstage and auxiliary propellant tank.

Table 5.2-1 Ground Test Objectives and Approach

TEST AREA	TEST HARDWARE	CONFIGURATION	TEST OBJECTIVES
STRUCTURAL TEST	STRUCTURAL TEST ARTICLE (STA)	COMPLETE OTV AND ASE STRUCTURE MASS SIMULATORS FOR SUB-SYSTEMS (AVIONICS, ELECTRICAL, RCS, ETC.) MECHANISMS	MODAL SURVEY LOADS STRUCTURAL LIFE DEPLOYMENT/RETRIEVAL SIMULATION
PROPULSION TEST	PROPULSION TEST ARTICLE (PTA)	STRUCTURE WITHOUT FORWARD BODY SHELL COMPLETE TANKS AND MAIN PROPULSION SYSTEM FOAM INSULATION ON TANKS AVIONICS RING WITH 75% OF AVIONICS	ENGINE START/STOP PROPELLANT LOADING TANK PRESSURIZATION TANK DEPLETION/PULL THROUGH PROPELLANT DUMP THRUST VECTOR CONTROL
FACILITIES	FACILITIES TEST VEHICLE	MAKE FROM STA COMPLETE FUNCTIONAL SUBSYSTEMS FOR BOTH STAGE AND ASE	KSC PHYSICAL AND FUNCTIONAL COMPATIBILITY TESTS CHECKOUT STATIONS AND GSE COMPATIBILITY MAINTAINABILITY TRANSPORTABILITY AND HANDLING
QUALIFICATION	FIRST FLIGHT VEHICLE	FIRST COMPLETE PRODUCTION VEHICLE FLIGHT TEST INSTRUMENTATION	ACOUSTIC THERMAL/VACUUM ELECTROMAGNETIC COMPATIBILITY
ELECTRICAL/ ELECTRONICS SYSTEMS INTEGRATION LABORATORY	FUNCTIONAL AVIONICS RING AND ASE AVIONICS	FUNCTIONAL AVIONICS RING ASE AVIONICS MODIFIED IUS SYSTEMS INTEGRATION LABORATORY	INTEGRATED AVIONICS SYSTEM COMPATIBILITY MISSION SIMULATION CHECKOUT STATION COMPATIBILITY SOFTWARE VERIFICATION

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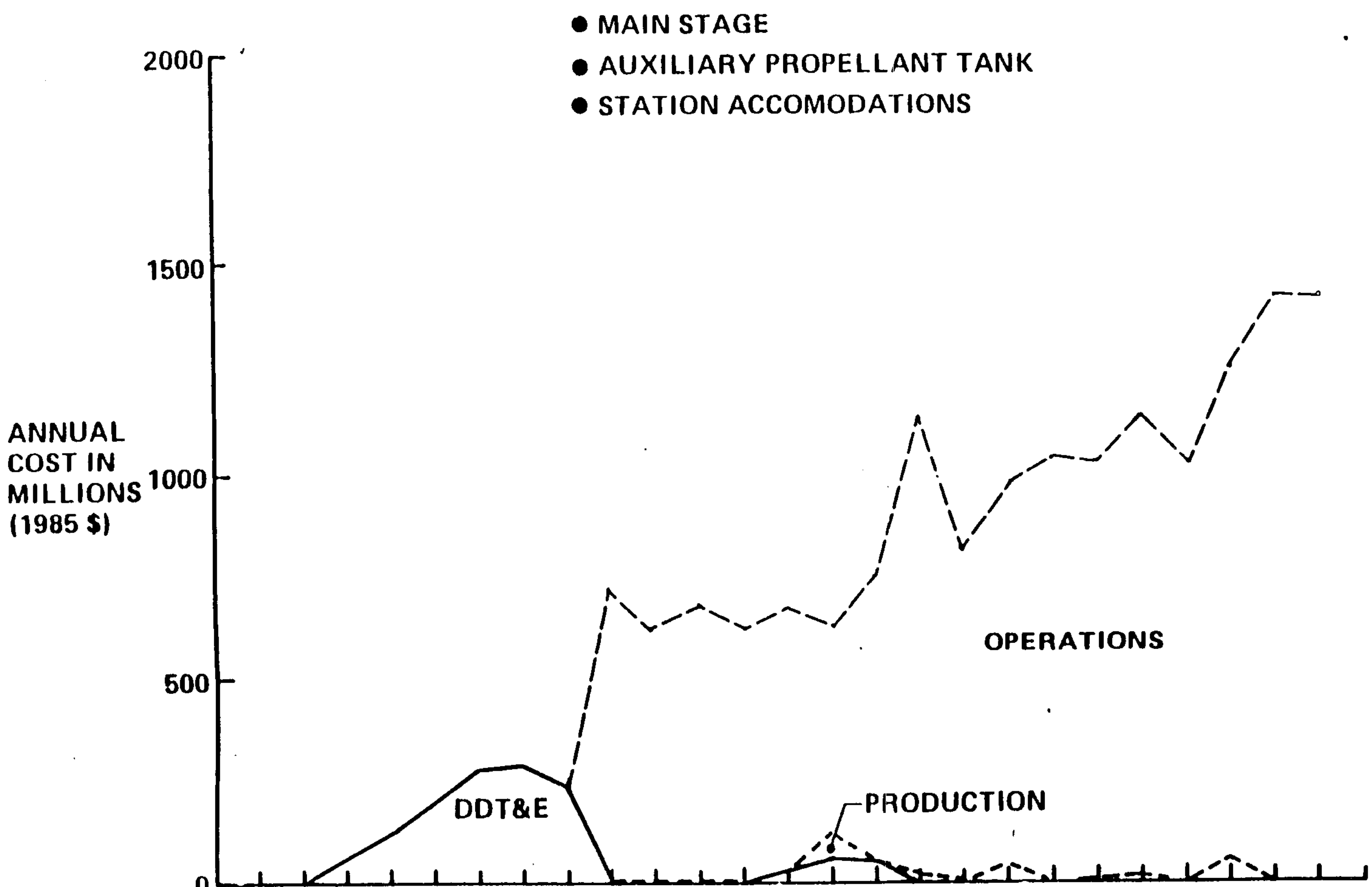

YEAR	86	87	88	89	90	91	92	93	94	95	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	TOTAL
DDT&E	0	0	65	120	204	270	288	222	0	0	0	0	29	63	56	0	0	0	0	0	0	0	0	0	0	$1317
PROD	0	0	0	0	0	0	0	0	0	0	7	0	0	63	0	20	7	50	0	7	13	0	57	0	0	$224
OPS	0	0	0	0	0	0	0	0	714	628	688	628	688	628	768	1101	814	978	1038	1019	1120	1039	1262	1407	1404	$15924

Figure 5.3-1 Funding Profile — Ground Based OTV Program

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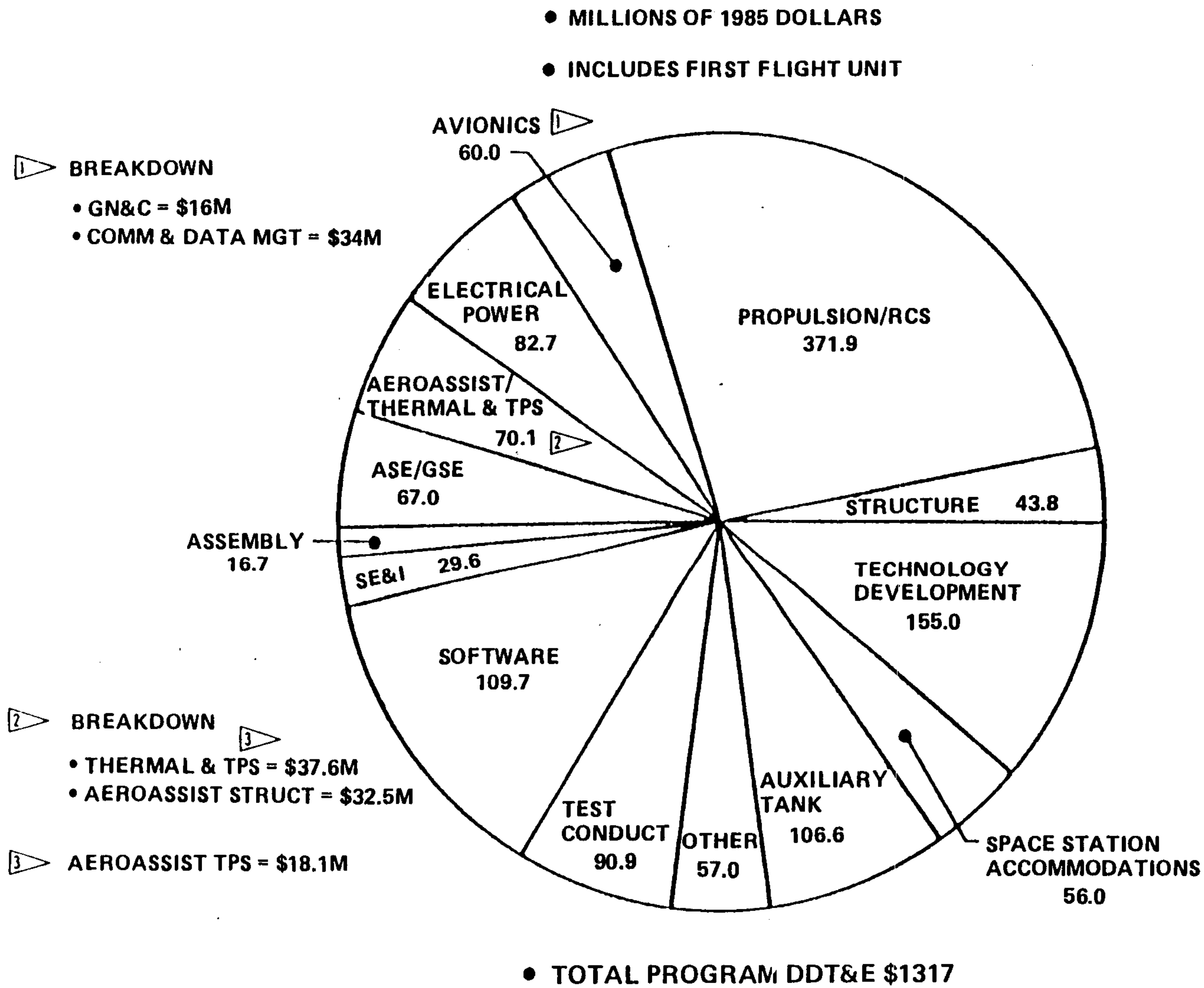

Figure 5.3-2 DDT&E Cost Breakdown – Ground Based OTV Program

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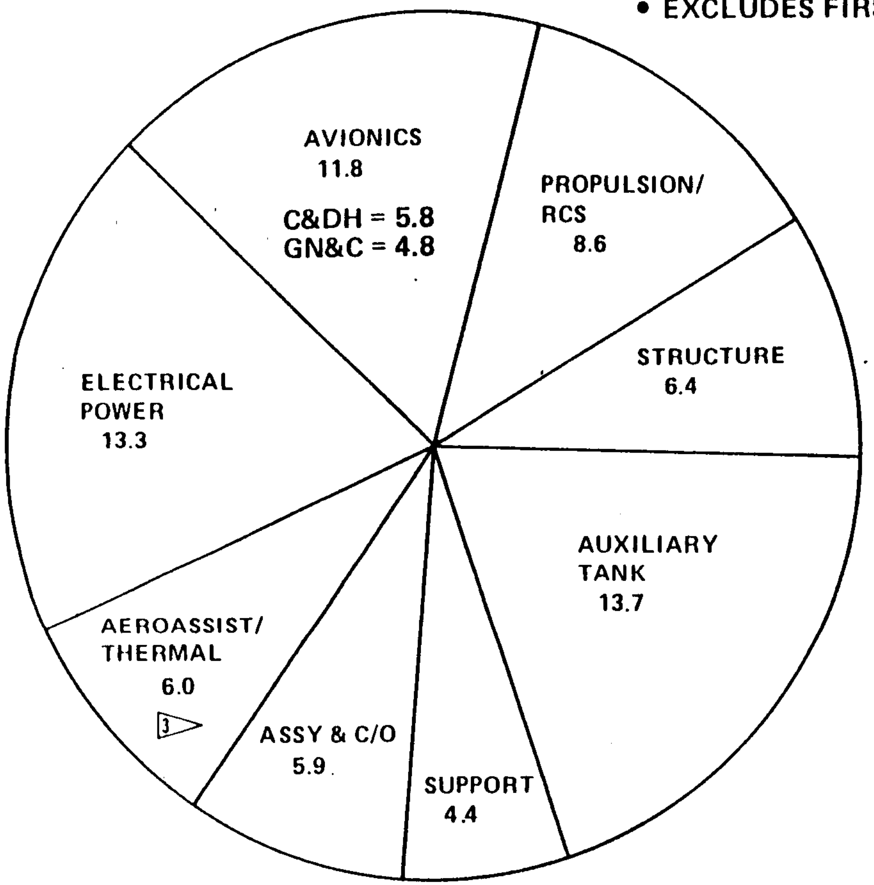
TFU = $70.1

- MILLIONS OF 1985 DOLLARS
- EXCLUDES FIRST FLIGHT UNIT



3 AEROASSIST = $3.5 (STRUCT)
THERMAL & TPS = $2.5

TOTAL PRODUCTION 1

ITEM	QTY	COST
STRUCTURE	3	$ 17.1M
ENGINES	14	34.7
OTHER PROPULSION	3	12.2
AVIONICS	3	31.5 [C&DH = 17, GN&C = 14.5]
ELECTRICAL POWER	3	35.5
AEROASSIST (HEAT SHIELD)	7	15.5 2
THERMAL	3	3.5
AUXILIARY TANK	2	26.0
ASSEMBLY & C/O	N/A	15.7
SUPPORT COST	N/A	11.7
S/S HANGAR	1	20.0
TOTAL		$223.4M

1 FIRST FLIGHT UNIT IN DDTE COST
2 BALLUTE INCLUDED IN OPS COST

- MAIN STAGE AVG UNIT COST = $50.1
- AUXILIARY TANK AVG. UNIT COST = $13.0

Figure 5.3-3 Production Cost Breakdown – Ground Based OTV Program

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Table 5.3-1 Cost Per Flight — Ground Based OTV Program

ITEM	GROUND-BASED OTV MAIN STAGE	GROUND-BASED OTV MAIN + AUX. TANK
TURNAROUND AND P/L MATING	1.5	1.6
OTV REFURB HARDWARE [3]	1.5	1.5
PAYLOAD INTEGRATION	1.5	1.5
OTV/PAYLOAD LAUNCH	73.0	135.0
FLIGHT OPERATIONS	----	0.9
GROUND OPERATIONS	0.5	0.6
ACCOMMODATIONS REFURB.	----	0.2
REFLIGHT COST	1.0	1.0
RELAUNCH ASE	----	0.4
TOTAL	79.0 [1]	142.7 [2]
APPLICABLE FLIGHTS	109	36

[1] WITH AMORTIZ. ADD $1.25M [2] WITH AMORTIZ. ADD $1.5M

[3] INCLUDES BALLUTE + AVIONICS BOX

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6.0 REFERENCES

1. Report No. D180-26090-1, Orbital Transfer Vehicle Concept Definition Study, Boeing Aerospace Company, Contract NAS8-33532, 1980.

2. Report No. GDC-ASP-80-012, Orbital Transfer Vehicle Concept Definition Study, General Dynamics Convair Division, Contract NAS8-35333, February 1981.

3. NASA Contractor Reports 3535 and 3536, Future Orbital Transfer Vehicle Technology Study, Boeing Aerospace Company, Contract NAS1-16088, May 1982.

4. Report No. GDC-SP-83-052, Definition of Technology Development Missions for Early Space Station, General Dynamics Convair Division, Contract NAS8-35039, June 1983.

5. Report No. D180-27979, Systems Technology Analysis of Aeroassisted Orbital Transfer Vehicle Low Lift/Drag (0-0.75), Boeing Aerospace Company, Contract NAS8-35095, 1985.

6. Final Report, Orbital Transfer Vehicle Launch Operations Study, Boeing Aerospace Operations, Contract NAS10-11165, January 1986.

7. Report No. D524-10005-3A1, Space Transportation Architecture Study, Interim Report Set III Vol. I, Boeing Aerospace, Contract F04701-85-C-0156, June 1986.